AFFTC-TR-53-33

PHASE II PERFORMANCE AND STABILITY TESTS OF THE YF-100A AIRPLANE

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ALFRED D. PHILLIPS
Project Engineer

FRANK K. EVEREST
Lieutenant Colonel, USAF
Project Pilot

December 1953

AIR FORCE FLIGHT TEST CENTER
EDWARDS AIR FORCE BASE, CALIFORNIA
AIR FORCE MATERIEL COMMAND
UNITED STATES AIR FORCE

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DATE 20 Jan 1954
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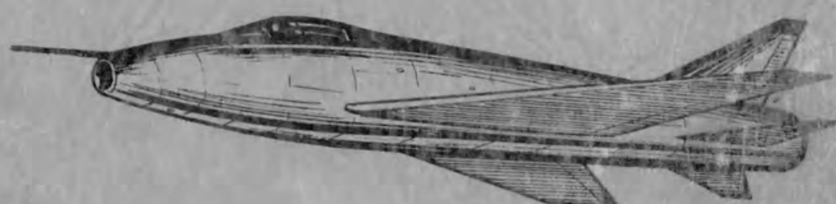
AIR FORCE FLIGHT TEST CENTER

AIR RESEARCH & DEVELOPMENT COMMAND



AF TECHNICAL REPORT
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TC RDS 4 MP-6-17-E, ...
AUTHORITY: AMC REPORT ...
DATE 15 Sep 58
BY:

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~~(OR CHANGED TO~~)
AUTHORITY: AMC /ARDC Sec class Guide 11 Feb 58
DATE 30 Oct 58
BY: Fy Stolike

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Wright Air Development Center

AIR RESEARCH AND DEVELOPMENT COMMAND

WRIGHT-PATTERSON AIR FORCE BASE,
OHIO

IN REPLY ADDRESS BOTH COMMUNICATIONS AND ENVELOPE TO COMMANDER, WRIGHT AIR DEVELOPMENT CENTER, ATTENTION FOLLOWING OFFICE SYMBOL:

WCOS

20 APR 1955

SUBJECT: (U) Phase II Performance and Stability Tests of the YF-100A Aircraft

TO: Commander
Air Force Flight Test Center
Edwards Air Force Base, California

1. Reference is made to WADC Letter dated 17 June 1954, Subj: Phase II Performance and Stability Test of the YF-100A AFFTC Technical Report No. 53-33.

2. The attached comments regarding recommendations contained in the referenced report are submitted for your consideration. The comments reflect the latest information available on the status and progress of corrective action to eliminate the reported deficiencies.

3. Paragraph references noted in the comments are the original headings contained in AFFTC Report No. 53-33, and are listed for convenience in referring to the report.

FOR THE COMMANDER

Frank W Williams

FRANK W. WILLIAMS
Lt. Colonel, USAF
Chief, Operational Services Division
DCS/Operations

1 Incl:
Reply to Phase Rpt
AFFTC-53-33

IF INCLOSURES ARE WITHDRAWN (OR NOT ATTACHED) THE CLASSIFICATION OF THIS CORRESPONDENCE WILL BE CHANGED OR CANCELLED IN ACCORDANCE WITH PAR 25E, AFR-205-1.

~~CLASSIFICATION CANCELLED~~
~~(OR CHANGED TO ________)~~
~~TT RDSMMR-6-17-E, 4 Jun 58~~
~~AUTHORITY: AMC REPORT NO. CD-356~~
~~DATE 15 Sep 58~~
~~BY:~~

CLASSIFICATION CANCELLED
~~(OR CHANGED TO~~ ________)
AUTHORITY: AMC/ARDC Sec Clas. Guide as rev 10 Oct 58
DATE 13 Jan 59
F. J. Stoliker

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55WCS-8914-A

55-2141

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~~CONFIDENTIAL~~

20 APR 1955

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AFFTC-53-33

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55WCS-8914-A

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

Reply to Phase Report ~~CONFIDENTIAL~~

AF Technical Report No. AFFTC-53-33

Par E, 1b - A mechanical stability device, referred to as a "Mach sensing stick force controller", was developed in conjunction with the original linear stabilizer control system to improve static longitudinal stability. The change from linear to non-linear stabilizer control system; however, has necessitated a redesign of this Mach box. The prototype of the redesigned Mach box will be made available to AFFTC on completion of present NAA tests.

Par E, 1c - A substantial improvement in low speed handling characteristics has been realized by the addition of the non-linear gearing system. Both the forces and stick motions necessary to control the airplane have been reduced for stabilizer trim positions used for landings. Increased aileron stick travel reduced sensitivity of lateral control and provided substantial improvement in low speed handling. Enlargement of the vertical tail, as a result of ECP F-100A-375, also contributed appreciably to the improvement of low speed handling characteristics.

Par E, 2a - This deficiency will be substantially reduced by the installation of the yaw damper on all F-100 aircraft. The production effectivity of this installation is F-100C, No. 164, with provisions made for retrofit of all inservice F-100 aircraft.

A prototype pitch damper is currently being flight tested by NAA and will be made available to AFFTC for testing as soon as the Contractor's tests are completed.

Par E, 2f - Pratt and Whitney has not provided a suitable configuration but this matter is still under study and it is anticipated that a final decision will be made in the next few weeks.

Par E, 2g - The stabilizer trim rate has been increased approximately 50%. This change in trim rate was achieved by increasing the speed of the trim motor. As a result of ECP F-100A-375, an impulse trim system has been installed in all F-100 aircraft which reduces the starting lag time thus producing an apparent increase in the trim rate.

Par E, 2h - There have been some minor changes initiated in the cockpit which have reduced the magnitude of this deficiency. Cognizance of this problem is maintained by the Laboratories and further improvement is anticipated with additional study.

Par E, 2k - A study has been made of an anti-skid system installation for the F-100A, F-100C, and F-100D airplanes. Approval has been granted for installation on F-100C, No. 164 and subsequent, and all F-100D airplanes. Retrofit of F-100A and C aircraft prior to No. 164 was disapproved by SMAMA, but is being further investigated by WADC.

Par E, 2l - There has been considerable improvement in the selection of drag chute configuration and mechanisms which have resulted in a high degree of reliability. The increase in reliability was accompanied by a slight decrease in actuating time; however, it is not anticipated that there will be any further reduction of time with the use of present equipment, which is considered to be the best available.

Par e, 2n - A substantial improvement in control response has been achieved by the modification required through implimentation of ECP F-100A-375. This modification included a redesign of the vertical tail and wing tip extensions and the installation of a non-linear gearing longitudinal control system. The merits and effects of this modification have been discussed elsewhere in this report and in WADC comments on Phase IV tests of the F-100A aircraft.

Par E, 2p - To correct this deficiency the bleed valves were staggered on production engines. In addition to this action, a study was initiated to consider the advantages of incorporating a modulated bleed system. This study is still current; however, the problem did not become as critical as it was anticipated. It is recognized that the staggered system may not be the optimum but it has proven to be acceptable.

Par E, 2q - It is believed that satisfactory and stall free engine operation to 50,000 feet altitude will be achieved in the near future. It has not been determined if the engine characteristics and/or fuel control system capabilities will permit totally stall free operation. J-57-P-21 engines with a redesigned compressor section are scheduled for F-100C No. 149 and subsequent.

Par E, 2r - The drag chute deployment handle has been relocated to a position on the shroud effective ship F-100A No. 154 and subsequent.

Par E, 2s - Refer to comment on Par E, 2b.

Par E, 2u - Flight experience has indicated that the F-100A does not have a pitch-up tendency as was originally anticipated during Phase II testing. No reports have been received by the WSPO indicating this characteristic is exhibited.

Par E, 1b - A mechanical stability device, referred to as a "Mach sensing stick force controller", was developed in conjunction with the original linear stabilizer control system to improve static longitudinal stability. The change from linear to non-linear stabilizer control system; however, has necessitated a redesign of this Mach box. The prototype of the redesigned Mach box will be made available to AFFTC on completion of present NAA tests.

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Comments on Phase II Performance and Stability Tests YF-100A

Par E.1.a - It is believed that this comment is closely associated with the comment on low speed characteristics. Pilots conducting recent preliminary APGC/TAC evaluation tests consider the over-nose visibility of the F-100A during take-off and landing as satisfactory provided that the aircraft is operated at the recommended speeds; that is, approach at 180 knots and touchdown at 150 knots for the landing weight. To provide increased visibility for the airplane the pilot's seat adjustment has been increased to allow a 1-1/4 inch additional upward travel. Also a 5° wedge block is being provided for use behind the parachute. This provides the pilot with a more favorable seat position. The seat changes are effective in production airplanes 3, 5, 6, 7, 16, 21, and subsequent.

The camera reticle glass has been widened and its supports have been moved outboard to lessen the interference of pilot's vision. This change has been made on all production airplanes. Also, all production airplanes embody glass in the side panels which reduces the optical distortions that were apparent on the prototype airplane.

Par E.1.b - The Contractor is aware of the stability deficit due to speed changes in the Mach number range between .85 and 1.0. As pointed out previously this condition is apparent in all aircraft operating through the transonic speed range and is caused by the normal aft movement of aerodynamic center. The apparent decay of stability is only noted as speed is changed and is not present in maneuvering flight. The Contractor has designed and is presently flying a Mach box. The purpose of this Mach box is to artifically build stable forces into the control system such that the pilot does not sense the unstable condition. Present tests to date show the Mach box fulfills all the requirements of pilot forces, hysteresis and eliminates the transonic stick force reversal with speed. Preliminary NAA pilot comments indicate that the Mach box improves the flying qualitites of the F-100A airplane in the Mach number range of .85 to 1.0, but the heavier stick forces and additional retrimming may have some disadvantage. Definite recommendations will be made as soon as more complete data and comments have been obtained.

Par E.1.c - The Contractor has installed an artificial stall warning device that shakes the rudder pedals and revised set of slats embodying a 15° rotation instead of the previous 10° on the F-100A aircraft. The results of these modifications have had a marked improvement as noted by the Company and Air Force pilots that have flown this configuration. The safe flight stall warning device and rudder pedal shaker has resulted in adequate stall warning for all low speed flight conditions, thus assuring that the pilot does not inadvertently enter the secondary stall. The revised slat configuration has provided better aileron control at the stall, improved longitudinal stability and has caused more uniform slat action than the original slat configuration.

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~~(OR CHANGED TO~~ 1)
BY AUTHORITY: AMC/ARDC Sec Clas Guide
DATE 9 Jan 58
BY: Fy Stoliber

CLASSIFICATION CANCELLED
~~(OR CHANGED TO~~)
TT RDSM AP 6-17-E 1 Jun 58
AUTHORITY: AMC REPORT NO. CD-356
DATE 15 Sep 58
BY:

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Other items that are currently under design and study are as follows:

1. Rudder Aileron Interconnect

This is being proposed to help eliminate the adverse yawing characteristics and increase the aileron effectiveness in accelerated stalls. This modification is now being flight tested.

2. Slat Snatcher

The slat snatcher is a method of retracting the outboard slats when the aileron is deflected up. It is mechanically so arranged that it will operate only in the low speed range. The purpose is to further improve the aileron effectiveness and essentially eliminate the adverse yawing under low speed accelerated stalls.

3. Enlarge Aileron Span

The purpose of this study is to increase the aileron effectiveness in the low speed flight regime at the expense of some loss in rolling performance at low altitude high speed. It is anticipated that tests on this configuration can be made by July.

4. Boundary Layer Control Flaps

Studies and wind tunnel tests of this arrangement indicate that a significant decrease in landing touchdown speeds may be made (approximately 20 - 25 knots). At the present time model tests are near completion. It is anticipated a full scale flight test model will be flying by October 1954. The reason that BLC flaps is very attractive to the low speed flying range is because not only is the angle of attack decreased and the lift increased, but also there is a large decrease in drag which will result in minimum rates of decent occurring at a lower indicated air speed, thus allowing decreased touchdown speed.

Par E.2.a - The yaw damper has been installed in production F-100A #2. Pilot comments to date are that dead beat damping through the speed range has been achieved. The pitch damper is presently undergoing laboratory tests on the flight simulator. Flight tests indicate that the yaw damper improves directional stability considerably. AFFTC evaluation of the pitch and yaw damper will be requested at completion of the pitch damper installation in the Contractor's aircraft.

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Par E.2.b - The breakout forces have been materially decreased by reducing the size of the main feel bungee, (to 3-1/2 #/degree - initial slope, 1#/degree - final slope) increasing the bob-weight size and deleting the valve centering spring. These changes are effective on all production aircraft. Pilot comments on the configuration to date have been very favorable.

In addition to the above, a nonlinear gearing between the stick and stabilizer similar in principle to that recently tested on the F-86D with such excellent results, is being designed.

Par E.2.c - With installation of the 15° slats which provide a measure of improved stability at low speeds and with the installation of the stall warning device (rudder pedal shaker), it is felt that the conditions surrounding this initial comment have been eliminated.

Par E.2.d - The Contractor has developed and tested a linear nose wheel steering system and it was found to be a definite improvement over the system initially installed. It has been released for installation on all production airplanes.

Par E.2.e - NAA pilots have made landings and take-offs with dive brake extended. No difficulty was encountered or excessive take-off distances encountered. Dive brake drag at take-off and traffic speeds is similiar to the F-86. Since complaints have not been raised against the lack of a light in the F-86, the WSPO has omitted this light. This was also done to simplify the aircraft, avoid excessive warning lights in the cockpit and divorce the aircraft from possible malfunctioning mechanical items not deemed essential.

Par E.2.f - Methods of accomplishing the intent of this comment have been under study. To date the best method has not been resolved. The most promising solutions are to locate provisions on the engine to actuate the nozzle position light already installed in the F-100A aircraft or to interlock the nozzle such that it cannot be open without afterburning. When those problems have been resolved, a proposal will be submitted.

Par E.2.g - The Contractor is currently pursuing a program to eliminate lag in the trim system. Some improvement has been obtained with the installation of the revised bungee and bob-weight and the deletion of the valve centering spring. Additional tests are scheduled to investigate the use of faster valves and improved trim mechanisms.

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Par E.2.h - Additional adjustments will be made to minimize the canopy reflection, as more night flying is accomplished by the Contractor.

Par E.2.i - In tests to date, the operation of the production engine in the F-100A airplane indicates that this condition is completely eliminated.

Par E.2.j - There are three methods of removing the canopy from inside the aircraft; (1) by normal electrical operation, (2) mechanically unlocking and raising canopy, and (3) blowing canopy off with emergency escape procedure. The canopy may be removed by external operation by two methods; (1) electrically canopy operation from exterior of the airplane, and (2) from exterior manual control unlock and manually raising the canopy. A parallel external blow-off control is considered dangerous due to inadvertent operation by unfamiliar ground personnel. No action taken on this deficiency.

Par E.2.k - During the course of the Contractor's flight testing, the wheel brake action has been found adequate; however, a number of mechanical difficulties have been experienced with the brakes. These problems are being taken care of by Bendix at this time.

The Contractor is further studying anti-skid braking devices which will allow the pilot to increase his braking effort without fear of skidding or destroying tires. A test installation will be made in the near future.

Par E.2.l - During the flight testing of the airplanes to date, reefed and non-reefed, parachute installations have been checked. The most successful parachute operations were made without reefing. The Contractor, AMC, and WADC have agreed to eliminate the reefing mechanism based on existing flight test experience. The pilot's drag chute deployment handle will be relocated to provide accessibility and simple operation.

Par E.2.m - The Contractor believes this comment arose from landings where some parachute panels failed. Subsequent testing did not cause any yawing to be apparent.

Par E.2.n - A number of studies are being pursued by the Contractor to improve response. The most notable are as follows:

Slat Snatcher
Aileron Rudder Interconnect
Extended Span Aileron

It is believed that the longitudinal control has improved and possibly the above comment is due, in some measure, to stick forces which subsequently have been improved by the addition of the revised bungee, bob-weight and elimination of the valve centering spring. In addition it is felt that they will be further improved by the revised variable gearing.

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Par E.2.o - Added complication for an automatic system does not warrent installation. Other aircraft have successfully operated without this feature. For simplication, this fix has been ommitted. At times for identification, pilots want to extend light yet keep gear up. No action taken.

Par E.2.p - Both the left and right bleed valves on the XJ57-P7 engine (YF-100A) operate simultaneously, which results in a rapid thrust change of approximately 500 pounds when the valves open or close. To reduce the rapid thrust changes on the J57-P-7 engine used in the F-100A, the left and right bleed valves operate at different engine speeds. This arrangement should give two small thrust changes during an engine speed advancement on the F-100A instead of the one large thrust change. The Contractor is in the process of flight evaluating a "Holley" governor which will greatly reduce the tolerance of actuation presently required of the bleed doors.

Par E.2.q - This problem has virtually been eliminated on the production J57-P-7 engine. However, a thorough and complete power control tests program will be conducted on the F-100A airplanes to discover and correct any control system deficiencies. It is anticipated that the fuel schedule of the Bendix control used on the F-100A may require some alteration to adapt it to the performance characteristics of the aircraft-engine combination. The fuel control testing and rescheduling program for production aircraft is normal for new installations. This work will be done after the "Holley" governor evaluation has been made.

Par E.2.r - A study is being made for the relocation of the drag chute release handle to the upper left hand instrument panel similar to the drag chute deployment mechanism on the F-86D. The deletion of the reefing mechanism greatly aids in making this installation possible.

Par E.2.s - A Mach box was developed by NAA and test flown. Flight test showed the hysterisis band to be too large. The test unit is being redesigned. Proper functioning Mach box should improve longitudinal stick-free stability. Final analysis of this unit's improvement on aircraft longitudinal stability is to be made by AFFTC in Phase IV test on the F-100.

Par E.2.t - This statement is caused by a basic misconcept of the slats on the F-100A airplane. These slats were designed primarily to improve the maneuverability, stability and limit lift coefficient at high Mach numbers. All indication to date show that these slats are operating essentially as they were designed to operate

The 15° slats give improved performance over the original 10° slats that were tested on the YF-100A airplane in that they have more uniform opening characteristics.

Par E.2.u - Existing tunnel data show that the 15° rotation slats give sufficient improvement in stability at the stall that very little pitchup tendency should be noted. This item will be flight investigated during the normal testing required.

Par E.2.v - The defrosting duct has been redesigned to give adequate distribution of cockpit airflow.

Par E.2.w - Gear-down limit speed has been increased to 250 K.

Par E.2.x - For pilot comfort the seat back angle has been changed from 17° to 12° by addition of a 5° wedge. Installation will be in aircraft Serial Nos. 758, 760, 761, 763, 771, 776, and subsequent.

Par E.2.y - A delta pressure type gauge has been developed for installation in all F-100 aircraft to provide adequate means of determining thrust. In event delivery of this gauge does not parallel aircraft production, space provisions will be left on the pilot's panel for retrofit.

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WCG

17 June 1954

SUBJECT: (U) Phase II Performance and Stability Tests of the YF-100A Aircraft AFFTC Technical Report No. AFFTC 53-33

TO: Commander
Air Force Flight Test Center
ATTN: DCS/Operations
Edwards Air Force Base
California

A study has been completed on the YF-100A final report, subject as above, on Performance and Stability Tests of the aircraft. Information as to action which has been or will be taken on the recommendations of the report is submitted herewith. Inclosure No. 1 is a list of remarks which are numbered to correspond to those sections of the report containing recommendations and conclusions.

1 Incl:
As stated above

ALBERT BOYD
Major General, USAF
Commander

IF INCLOSURES ARE WITHDRAWN (OR NOT ATTACHED) THE CLASSIFICATION OF THIS CORRESPONDENCE WILL BE CANCELLED IN ACCORDANCE WITH PAR. 25E, AFR-205-1.

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AF Technical Report No. AFFTC 53-33
December 1953

PHASE II FLIGHT TEST OF THE NORTH AMERICAN YF-100A AIRPLANE USAF No. 52-5754

ALFRED D. PHILLIPS
Flight Test Engineer

FRANK K. EVEREST, Lt. Col., USAF
Test Pilot

UNITED STATES AIR FORCE
AIR RESEARCH AND DEVELOPMENT COMMAND
AIR FORCE FLIGHT TEST CENTER
EDWARDS, CALIFORNIA

PUBLICATION REVIEW

This Report has been reviewed and is approved

H. A. Hanes

H. A. HANES
Colonel, USAF
Director, Flight Test and Development

J. S. Holtoner

J. S. HOLTONER
Brigadier General, USAF
Commander

ABSTRACT

This report presents the results of Phase II performance and stability tests of the North American YF-100A air superiority fighter. The test results indicate the aircraft is superior in performance to any production fighter in the USAF. The most serious defects of the aircraft are the inadequate visibility over the nose during take-off and landing, the poor low-speed handling characteristics, and the negative to neutral static longitudinal stability experienced in level flight from approximately .8 Mach to maximum level flight speed.

TABLE OF CONTENTS

A. PURPOSE

This report presents the results of limited performance and stability flight tests of the YF-100A airplane, USAF No. 52-5754.

B. INTRODUCTION

1. Extent and Location of Flight Tests:

a. Flight tests of the aircraft were conducted to obtain stability and performance data to determine the aircraft's handling qualities and to compare actual performance with contractor's estimated performance.

b. The Phase II test program on the YF-100A airplane was conducted at the Air Force Flight Test Center, Edwards, California from 3 September through 17 September 1953. Thirty-nine flights, totaling 19 hours and 42 minutes, were required to complete the limited performance and stability tests.

2. Description of the Aircraft:

a. The YF-100A is the prototype version of the F-100A aircraft, an entirely new design of a single-place fighter capable of tactical operation in the high transonic and low supersonic speed range. The airplane is characterized by a thin, highly swept (45°), moderate aspect ratio wing, low drag fuselage and canopy, inboard ailerons (with no flaps), automatic leading edge slats, low positioned, one-piece horizontal stabilizer, and a thin-lip engine air inlet. The landing gear is of the tricycle type with the main gear retracting inboard into the wing and fuselage and the nose gear retracting aft into the nose section of the fuselage. Other noteworthy features of the airplane include a fuselage speed brake located in the bottom of the fuselage in the wing center section and an irreversible power control system with artificial feel.

b. The aircraft is powered by a single Pratt and Whitney XJ-57-P-7 continuous flow axial type gas turbine with afterburner rated at 13,800 lbs thrust uninstalled maximum power. The engine has a split compressor, split turbine, a hydro-electronic fuel control and is equipped with a two-position exhaust nozzle. The two spools of the compressor and turbine are mechanically independent of each other.

3. Test Configuration:

The aircraft was tested in the clean configuration and with two empty 275 gallon external fuel tanks attached to the undersides of the wings. The test weight of the aircraft was found to be 25,370 pounds in the clean configuration with fuel and oil tanks full. This weight included all test instrumentation, 1289 pounds ballast and the pilot. The aircraft was loaded to obtain a mid cg position and all tests were made at this one loading. The cg during flight was not controlled as the aircraft utilized automatic fuel sequencing. The cg position varied from a most aft position of 31.7 per cent MAC to a most forward position of 29.5 per cent MAC as fuel was consumed during flight. Detailed weight and balance data and photographs of the test aircraft are presented in Appendix II.

C. SUMMARY OF TEST RESULTS

1. Performance:

a. The results of the performance tests of the YF-100A type aircraft indicate that it is superior to any production aircraft in the United States at this time. The YF-100A can take-off at 25,000 pounds gross weight using 3000 feet of runway on a standard day at sea level. The aircraft can then climb at a rate of 14,800 feet per minute (at sea level) and reach an altitude of 45,000 feet in 4.3 minutes from best climb speed at sea level. Approximately two minutes must be added to obtain time to take-off and accelerate to best climb speed. The YF-100A aircraft can continue to climb to a combat ceiling (500 feet per minute) of 53,000 feet. With afterburner off, the climb is very slow (16 minutes from sea level to 40,000 feet). It is felt that climbs without afterburner will not be utilized by tactical organizations to a very large degree. At 35,000 feet a level flight Mach number of 1.04 is attainable. Maximum Mach number obtained was 1.39 indicated in a dive from 51,000 to 20,000 feet. There is no wing roll-off as previously experienced in F-86 type aircraft, no high Mach number buffet, no objectionable trim changes or other noticeable compressibility effects. Rates of descent reached approximately 50,000 feet per minute. This airplane is strictly a hit-and-run type fighter when utilized against present day fighters; i.e., it should not be expected to fight with an F-86 at the F-86's speed. This condition is comparable to the F-80 vs the F-51. Comparative performance between the YF-100A and other present day fighters is presented below:

COMPARATIVE PERFORMANCE

	YF-100A Test Results Clean	F-86E Memo Report FTD 51-19 Clean	F-86D Ref A F Tech Rpt 53-26 Clean	F-94C Ref A F Tech Rpt 53-30 Tip Tanks	F-94B Ref A F Tech Rpt 52-26 Tip Tanks	F-89A Memo Report FTR 51-13 Tip Tanks
Max. Mach No. (max. power)	1.04 35,000 ft	.90 25,000 ft	.923 35,000 ft	.835 35,000 ft	.785 35,000 ft	.874 25,000 ft
Rate of climb at S. L. (ft/min)(max. power)	14,800	8,800	12,900	9,500	7,100	7,700
Rate of climb at 30,000 ft (max. power)	10,500	2,850	4,900	4,500	3,000	4,900
Combat ceiling (max. power) 500 ft/min.	53,000	46,000	47,000	47,000	43,000	39,500

COMPARATIVE PERFORMANCE
(Continued)

	YF-100A Test Results Clean	F-86E Memo Report FTD 51-19 Clean	F-86D Ref A F Tech Rept 53-26 Clean	F-94C Ref A F Tech Rpt 53-30 Tip Tanks	F-94B Ref A F Tech Rpt 52-26 Tip Tanks	F-89A Memo Report FTR 51-13 Tip Tanks
Time to Climb to 45,000 ft min (max. power)	4.3	15.5	9.3	10.5		

b. Engine performance and reliability are considered excellent. The afterburner is especially reliable inasmuch as afterburner "lights" can consistently be made at any altitude, speed, or configuration. Three objectionable characteristics associated with engine operation are: (1) slow, sluggish engine response to control application, (2) the opening and closing of compressor surge bleed valves when operating at power settings of approximately 90 percent rpm, which cause sudden increase and decrease in thrust, (3) compressor stall as occurred when applying power at low power settings (70 percent rpm).

2. Stability and Handling Characteristics:

a. The airplane is easy to taxi with the exception of high rudder forces. Take-off is satisfactory except at lift-off when visibility is limited. The airplane tends to wallow somewhat due to low airplane response to stabilizer movement. The airplane must be "pulled off" the runway as it exhibits no inclination to fly itself off even at speeds up to 180 knots. The visibility is not satisfactory during climb because of the nose high attitude at best climb speed.

b. The airplane exhibits negative to neutral static longitudinal stability from approximately .8 Mach number to maximum Mach number. Trimming the aircraft for normal cruise is very difficult because of this factor. It is very annoying for formation flying and would possibly be detrimental to the airplane as a gun platform. In the approach configuration, the static longitudinal stability is positive; however, it is felt that it could possibly be increased and low speed handling characteristics could be improved. The YF-100A airplane exhibits a lateral, directional, and longitudinal poorly damped dynamic stability characteristic which occurs at high altitude and high Mach. The longitudinal damping improves with decreased altitude but lateral and directional damping is unsatisfactory at all altitudes. This, too, is detrimental for formation flying and accurate gun firing. Longitudinal control is sensitive, and at transonic speeds of approximately .8 to .9 Mach number, it is similar to that exhibited by the F-86D aircraft. This is unsatisfactory, however, it is felt that pilots could live with this condition.

c. The speed brake is very effective and creates a considerable amount of drag with little objectionable buffeting. The speed brake may be used on landing; however, because of the small buffet encountered at full extension, it gives the pilot a false impression that the airplane may be nearing the stall.

d. The stalling characteristics are poor. At 10,000 feet, from 160 knots to 110 knots, there is a yawing and pitching tendency that is quite uncomfortable to the pilot. At 110 knots the left wing drops uncontrollably; full right ailerons and rudder not being capable of controlling the wing drop. The stall itself is unsatisfactory because of this drop but is considered acceptable because the normal touchdown speed is well above the stall speed.

e. Landing of the YF-100A airplane is considered more difficult than any other production fighter. It exhibits poor lateral and longitudinal low speed handling characteristics. The airplane response to control movement is slow and the pilot finds himself constantly over-controlling during final approach, both because of the high stabilizer breakout forces and the low airplane response rates. During the latter part of the final approach, the nose is very high and it is extremely difficult to judge the altitude and position of the airplane when nearing touchdown point. When landing at night the limited visibility makes landings dangerous inasmuch as the pilot has little or no reference to either side as he does during daylight hours. On one occasion during a landing, the pilot landed 2000 feet short of the runway as a result of the high rate of sink and poor visibility. Touchdown speeds are high because of the ease of dragging the tail. On one landing the tail touched the runway at a touchdown speed of 136 knots. Normal touchdown speeds will be approximately 150 knots or above. It is anticipated that all landings will be made utilizing the drag chute. The drag chute is unacceptable at this time owing to the considerable amount of yaw it creates after deployment.

D. CONCLUSIONS:

It is concluded that:

a. The YF-100A series airplane out-performs any other production fighter in the USAF at this time.

b. The YF-100A airplane is unacceptable for service use at this time because of poor visibility over the nose during take-off and landing, neutral and negative static longitudinal stability, and the poor low-speed handling characteristics.

c. The YF-100A aircraft has a good potential and can be made into a useful tactical weapon.

E. RECOMMENDATIONS:

1. The following recommendations pertain to major deficiencies which should be changed prior to acceptance or delivery of the aircraft to tactical units. It is recommended that:

a. Forward visibility be increased and improved during take-off and landing.

b. The neutral and negative static longitudinal stability exhibited be corrected to insure ease of formation flying and to increase the potential of the airplane as a gun platform.

c. The poor handling characteristics during take-off and landing configuration be improved.

2. The following recommendations pertain to deficiencies which are considered unsatisfactory and should be corrected as soon as possible. It is recommended that:

a. The lateral-directional and longitudinal dynamic stability be improved.

b. The stabilizer breakout forces be decreased.

c. The poor stalling characteristics be improved.

d. The rudder forces during taxiing and low-speed flight be decreased.

e. A dive brake warning light or position indicator be installed on the pilot's panel to prevent inadvertent take-off with dive brakes extended.

f. An afterburner light be installed to give indication that the afterburner is on.

g. The stabilizer trim response rate be increased.

h. The canopy reflections of cockpit instrument and flood lights be eliminated for night flying.

i. The slow, sluggish engine response to control application be eliminated.

j. A switch be installed on the exterior of the fuselage so that the canopy may be blown off by external operation on the ground.

k. The wheel braking action be increased.

l. The time of actuation of the drag chute be decreased.

m. The yawing moment caused by drag chute deployment be eliminated.

n. The rudder, aileron, and stabilizer airplane response rates at low speeds be increased.

o. An automatic landing light retraction be coupled with gear retractors to prevent inadvertent flight with the landing light extended.

p. The sudden increase or decrease of thrust during actuation of the surge bleed valves be eliminated.

q. The compressor stall experienced at low power settings be eliminated.

r. The drag chute deployment handle be relocated to a position behind the throttle.

s. A study be made to determine the feasibility of installation of a "q" device for better aerodynamic longitudinal control forces throughout the entire altitude and speed range.

t. The slats be modified to insure that they do not extend in unaccelerated flight above .9 Mach number at high altitude.

u. A study be made to insure that the pitch-up tendency will not cause inadvertent over-stressing of the airplane at low altitude, and high speed when making abrupt control movements.

v. The defrosting duct presently under the left console be relocated to prevent excess heating of the left console.

w. The limit gear down speed be increased to 250 knots IAS.

x. The pilot seat tilt back angle be reduced to approximately 7 to 9° for pilot comfort.

y. An adequate means of determining thrust during take-off and during flight be provided by an engine pressure ratio type gage.

APPENDIX I
FACTUAL DATA
YF-100A USAF No. 52-5754

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(continued)

A. PILOT'S COMMENTS

1. Cockpit Description:

a. Access to the cockpit is by use of a ladder that hooks over the edge of the cockpit. There are foot steps and handholds installed in the side of the fuselage; however, they are too high to be utilized without the use of an extension ladder to begin the initial climb up the side of the fuselage. Generally, the cockpit is considered acceptable; however, the following discrepancies should be corrected in the production aircraft:

b. The lever for drag chute deployment is located directly outboard of the throttle. It is quite awkward to manipulate, especially when the throttle is in the retarded or low power position. The drag chute deployment handle should be relocated to a position directly behind the throttle.

c. The windshield defrosting handle is too close to the landing gear handle when the gear is down. The windshield defrost handle should be relocated approximately one inch or more outboard from its present position and for ease of operation, the landing gear handle should also be relocated or extended 0.5 inches outboard from its present position.

d. The pilot's seat is presently too difficult to move up. A stronger spring or bungee should be installed for ease of operation.

e. The map case cover opens inboard and should be changed to lift up and outboard for ease of operation.

f. The present warning lights located on the left console should be relocated up and more in line with the pilot's vision, such as those presented at the F-100 Contractor Technical Compliance Board. On several occasions the pilot had the oil overheat warning light come on and did not notice it until after some time had elapsed.

g. It is too difficult to read the position indicators for the rudder pedal adjustment. The rudder pedals should be made adjustable fore and aft with one control instead of individual controls.

h. Visibility from the cockpit is considered excellent in the taxi and level flight attitudes, and visibility inside the cockpit is considered satisfactory with the exception of the above mentioned warning lights. (See page 24, Appendix II for photographs of the cockpit).

2. Taxiing and Ground Handling:

The YF-100A Aircraft is fairly simple to taxi and is considered similar to the F-86 type aircraft in ground handling. One difficulty noticed is the high rudder pedal forces when utilizing nose wheel steering. This makes the use of nose wheel steering somewhat difficult and the pilot finds himself prone to use the brakes more than is necessary. The nose wheel steering ratio appears to be satisfactory during both low and high taxi speeds. The YF-100A

aircraft was taxied in cross winds up to 20 knots with little or no difficulty. Braking action during taxiing is considered good. The brake pedal forces are light and are considered far superior to those of the F-86 type aircraft. During taxiing the rudder is of little or no use and brakes and/or nose wheel steering must be used at all times.

3. Take-Off and Initial Climb:

Take-offs were made with full power (afterburner on, speed brakes in). The brakes were released when the afterburner cut in and nose wheel steering was used up to approximately 100 knots IAS. Rudder control is available at approximately 60 knots; however, the response of the airplane to the rudder is slow and it is recommended that nose wheel steering be utilized up to approximately 100 knots. Inasmuch as the horizontal stabilizer, if held in full nose up position, creates quite a bit of drag, the stick is left at the take-off trim position (approximately 4 degrees nose up) until an indicated airspeed of 140 knots is obtained. At this time the stick is pulled firmly to the full nose up position and the airplane is flown off the runway at 150 knots IAS. Because of the unusually high angle of attack required for take-off with the highly swept, low aspect ratio wing with no flaps, a conscious effort must be made to pull the aircraft off the ground. On one occasion, an indicated airspeed of 180 knots was attained on the ground when the nose wheel was held approximately 6 to 12 inches off the ground and it was still necessary to increase the pitch attitude for take-off. After the airplane broke ground, the control stick back pressure was released slightly and the gear was retracted immediately. There is little or no trim change during gear retraction; however, the pilot found that there is a slight tendency to overcontrol laterally because of the light aileron breakout forces. Take-off is considered satisfactory; but, because of the above mentioned directional problem of the high rudder forces and low airplane response to rudder, it appears that formation take-offs will be difficult. During the transition time from lift-off to gear retraction, there is little forward visibility because of the nose high attitude of the airplane. This is not considered unsatisfactory, however, because the transition time between lift-off and gear retraction is short and the pilot can level the airplane out almost immediately after lift-off to increase the speed to the recommended climb speed. The longitudinal stick forces are considered too high and this, added to the fact that immediately after breaking ground the response of the airplane to longitudinal control position is slow, may result in overcontrolling slightly. This is not considered satisfactory; however, it is acceptable because of the previously mentioned rapid acceleration of the airplane during the transition period to higher speeds where longitudinal control is satisfactory.

4. Climb:

After take-off the YF-100A aircraft accelerates very rapidly to the best climb speed and visibility is considered acceptable during this part of the flight. Once the recommended climb speed is attained, the nose is pulled up sharply and the YF-100A climbs very rapidly to its combat ceiling. The airplane has an extremely nose high attitude during climb and visibility is not considered satisfactory. No unusual or unsatisfactory control characteristics are evident during the climb, but care must be exercised to prevent the airplane from exceeding the best climb speed, inasmuch as the nose high attitude and rapid rate of climb

combine to give the pilot a tendency to level out for better visibility.

5. Level Flight:

The maximum level flight Mach number attained was approximately 1.04 at 35,000 feet and the maximum level flight indicated airspeed attained at sea level was 631 knots (.96 Mach no.). Aerodynamic characteristics are not considered satisfactory inasmuch as there is a slight yawing and ptiching moment that makes formation flying difficult and also makes the airplane a poor gun platform. Further, the longitudinal breakout forces are considered high. These, coupled with negative longitudinal static stability throughout the cruising range, cause the pilot to be constantly retrimming and moving the stick in an attempt to fly the airplane at a constant altitude. The cockpit pressurization is excellent. Cockpit refrigeration is inadequate during low level, high speed flight as the pilot is too warm even when dressed in a light flying suit. It is believed that part of this heat results from the defroster duct that runs along the left console since, it is not possible to touch any part of the left console with the exception of the throttle due to the high temperature of the metal parts. Some consideration should be given to relocating the defroster-heater duct and/or control valve to keep this temperature problem to a minimum. The canopy fogged over at times when at extreme altitudes but it cleared rapidly upon actuation of the de-fogging system. The seat is comfortable but it is tilted back too much and should be straightened possibly three to five degrees.

6. Flight Control System:

a. All three flight control systems are irreversible and are powered by two engine driven hydraulic pumps, one taken off each rotor section. There is an emergency air driven hydraulic pump which was never utilized, and it is not known whether it would be usable in case of emergency or not. Control "feel" is obtained by use of bungees at the control stick and on the rudder. In the case of the stabilizer control, there is also a bob-weight. There is no aerodynamic or "q" device installed that decreases or increases the stick forces with increase or decrease of airspeed and/or altitude. Incorporation of a "q" device would improve the control "feel". Stabilizer breakout forces are considered too high. Lateral control forces are considered satisfactory at all speeds. Rudder forces are considered acceptable at high and intermediate speeds, but are too high at the approach and take-off speeds. Lateral and longitudinal trim is obtained by use of the trim button located on top of the control stick and there is a toggle switch on the left console to actuate rudder trim. Aileron and rudder trim are satisfactory and the stabilizer trim speed is also considered satisfactory; however, stabilizer trim motor response is too slow and is considered unsatisfactory.

b. Longitudinal Stability

(1) Stabilizer control and effectiveness for take-off.

Stabilizer control during take-off is considered satisfactory. It was possible to lift the nose wheel from the ground between 110 and 120 knots IAS.

The airplane was flown only at one mid. cg (32 percent), however, it was felt that, inasmuch as the airplane lifts off at 150 knots, the stabilizer control effectiveness would be satisfactory even at a forward cg. As mentioned previously immediately after breaking ground there is a tendency to over-control because of the high breakout forces and low response rate. If possible, the high breakout forces should be eliminated inasmuch as some pilots might have difficulty during take-off in turbulent and/or cross wind conditions. There is little or no trim change during the transition period from lift-off during gear retraction until recommended climb speed is attained.

(2) Dynamic stability.

Longitudinal dynamic stability is satisfactory in the low and intermediate speed ranges. At high Mach number (.98 and above) and high altitude (45,000 feet) the controls free tests indicate marginal dynamic stability as it takes 4 to 5 cycles before a longitudinal oscillation dampens. When an attempt is made to stop the longitudinal oscillations by use of the controls the pilot tends to over-control, similar to the F-86F and F-86D models. This condition is considered unsatisfactory. The sensitivity of the controls makes it very easy to overshoot the desired acceleration in maneuvering flight. Attempts to correct the "overshoot" usually cause the pilot to induce a longitudinal oscillation similar to that encountered in the F-86D aircraft. This condtiion is not as severe as that experienced in the F-86D; however, it does warrant some consideration for correction because this condition is unacceptable when flying formation and in rough air.

(3) Maneuvering stability.

Maneuvering stability is considered to be unsatisfactory. It is felt that at a high indicated airspeed it would not be too difficult to exceed the limit load factor by abrupt displacement of the control stick because of the inherent pitch-up tendency experienced in swept wing fighters. This pitch-up is more noticeable than that of earlier swept wing fighters, such as the F-86 aircraft, and it is considered that this is due to the increased wing sweep angle. When maneuvering the YF-100A aircraft, upon application of any "g" force, the drag increase becomes quite apparent and it may be determined now that this fighter can only be utilized as a hit and run gun platform as any attempts to linger and try to out-turn any other present day fighter would result in a great loss of speed. This, coupled with poor turning capabilities at lower speeds, would mean that the opponent would invariably end up in a stern attack on the YF-100A aircraft. As long as the YF-100A is kept at high speeds and low "g" factors, it should have a very good potential as a fighter. During one phase of the test program, at an indicated Mach number of 1.1 and at 35,000 feet altitude, the stick was pulled full back. A maximum acceleration of 5.6 "g's" was obtained. The Mach number decreased to less than 1.0. A pitch-up was noticed at 5 "g's" which resulted in an acceleration of 5.6 "g's" even though the pilot was attempting to hold a constant 5 "g" pull up. This may be an indication of the difficulty to be expected at lower altitudes where more "g's" may be obtained from abrupt application of the stick and over-stressing the airplane may be possible. Further study is recommended to insure that the airplane will not be easily over-stressed during accelerated maneuvers at medium to low altitudes and at high airspeed.

(4) Static Stability

Static stability of the YF-100A aircraft is considered unsatisfactory. At cruising and high Mach numbers the YF-100A exhibits negative stability. When increasing speed from a pre-determined trim speed, stick position and forces required to maintain the slower speeds are in the wrong direction (pull), and when decreasing speed from a pre-determined trim speed, the stick position and force changes indicate neutral to very slight positive stability. This is considered an unacceptable condition and every effort should be made to correct this condition prior to delivery of production airplanes. In the approach configuration the airplane exhibits positive static stability; however, because of high breakout forces it is difficult to fly in a trim condition. The unsatisfactory static stability tends to make the airplane difficult to fly in formation and results in a poor gun platform.

(5) Trim Change Characteristics

There is a slight trim change when extending the gear, as the left gear apparently extends first, causing a yawing moment to the left; however, it is easily controlled by use of the right rudder. When extending speed brakes at high speeds there is little noticeable trim change and this is considered excellent. There is approximately a five to ten pound nose down trim change when igniting the afterburner at the intermediate to higher speed ranges. This is also controllable and is considered acceptable. If possible, it is recommended that this trim change be alleviated, since in the case of formation flying, it would present a small problem.

(6) Stabilizer Control and Effectiveness for Landing

Stabilizer control and effectiveness for landing is considered satisfactory; however, the response of the airplane to control movement is considered unsatisfactory. Coupled with the high breakout forces and low response rates, the pilot finds himself constantly overshooting his desired attitude during the approach and landing flare. No trouble exists on control effectiveness inasmuch as it is quite easy to drag the tail during touchdown. The tail was dragged on one landing when touchdown was made at an indicated speed at 136 knots. Most touchdowns were made at approximately 150 knots IAS. The stabilizer breakout forces should be reduced and a study should be made to increase airplane response to control movement during the approach configuration. Perhaps the control effectiveness would be improved by return to the old type elevator control linkage, similar to that of the F-86E, for slow speed operation.

(7) Stalls

Stalls were made only at the mid cg position. The stalling characteristics are considered poor. Several full stick back stalls were made at both low and high altitudes. At low altitudes (10,000 ft.) from approximately 160 knots down to the stall of 110 knots there is a yawing and pitching tendency that is quite uncomfortable to the pilot. At 110 knots the left wing drops uncontrollably inasmuch as full right aileron and full right rudder will not hold the wings level. The stall itself, when reached at approximately 110 knots, is

unsatisfactory but is considered acceptable because there is no reason why the airplane will ever be flown at this low speed, especially as the normal touch-down speed will be 150 knots or above. Stall recovery is made by releasing back pressure on the stick while still holding full right aileron and right rudder; thus permitting the speed to increase until such time as the aileron and rudder effectiveness return. Accelerated stalls are considered satisfactory inasmuch as the airplane buffets and pitches similar to other swept wing aircraft and stall recovery is made by releasing back pressure on the control stick. At high altitudes (35,000 ft) during unaccelerated stalls the stick can be held back to full aft position and although the airplane is prone to pitch and roll, there is enough aileron and rudder control to rudder-walk the airplane through loss of several thousand feet of altitude.

c. Directional stability

(1) Cross wind take-off Characteristics.

The cross wind take-off characteristics of the YF-100A are considered satisfactory inasmuch as, with the nose wheel left on the ground to 140 knots, there is little or no tendency for the upwind wing to rise. Directionally the airplane has a tendency to turn into the wind and the normal procedure is to correct with nose wheel steering. The best procedure is to use nose wheel steering to correct for turning tendencies up to higher indicated speeds as the velocity of the cross wind is increased. Brakes could be used for directional control. However, brake response is quite sensitive and the pilot is somewhat prone to over-control; thus, it is best if the nose wheel steering is relied upon for directional control during cross wind take-offs.

(2) Dynamic lateral-directional Stability.

Dynamic lateral-directional stability is considered unsatisfactory throughout the entire combat speed range. Directional-lateral oscillations, resulting from rudder displacements, are not damped after 5 cycles and show no tendency to damp at all during slightly turbulent air conditions. This is considered unacceptable and should be corrected prior to acceptance or delivery of the aircraft to tactical units.

(3) Static directional Stability.

Static directional stability of the YF-100A aircraft is considered satisfactory at all speeds. The rudder and aileron position and forces are in the right direction for a given yawed condition. At airspeeds of 160 knots and below, with full right rudder, the airplane will roll to the right even though full left aileron is applied. This is undesirable but acceptable because full rudder sideslips need not be made in this airplane during the approach and landing flare. The rudder forces are satisfactory in the high and intermediate speed ranges, but during approach speeds the rudder forces are too high, and coupled with the low airplane response to the rudder, rudder control is unsatisfactory.

d. Lateral Control

The lateral control of the YF-100A aircraft is satisfactory. The rate of roll is high and the time to reach peak rate of roll is satisfactory at all speeds down to approximately 175 knots. From 175 knots down to the touchdown speed of 150 knots, the rate of roll is satisfactory, but the response to aileron deflection is slow and some difficulty may be experienced during turbulent and/ or cross wind landing conditions. There is a slight amount of adverse yaw present in the lower speed range; however, it is considered acceptable inasmuch as this speed is below the practical combat speed of the airplane.

e. Dive and High Speed Flight Characteristics

High speed dives were made up to an indicated mach number of 1.39. During the dives there was no rolling tendency such as that experienced in F-86 type aircraft and the YF-100A exhibits the best diving characteristics of any fighter flown to date. Recovery from dives can be made by either actuating the speed brakes, pulling power off and slowing down, or by pulling out at the desired altitude. Again, caution must be exercised when making high "g" pull-outs because of light stick forces necessary to recover from high speed dives. As previously mentioned, it is believed that it would be quite easy to over-stress the airplane if caution were not exercised during high speed flight. One dive was made with drop tanks attached (empty) to an indicated Mach number of 1.2. There is no perceptible difference in handling characteristics of the airplane. There is no buffet exhibited with the tanks installed and the installation is considered to be superior to that on any previous model or type.

7. Approach and Landing:

Approach and landing of the YF-100A aircraft is difficult and at night is considered to be dangerous. Initial approach is made at an altitude of approximately 1000 ft. above the runway. Indicated airspeed at peel-off is between 300 and 350 knots and, at the time of peel-off, speed brake is extended and power reduced to 80 percent. As the airplane slows down to gear extension speed of 210 knots, the gear is extended and the speed brake is retracted. The gear limit speed of 210 knots is too low and should be increased to 250 knots. Speed during base leg and turn onto final approach is held at 170-180 knots and, after roll-out onto final is made, 170 knots is held until approximately 1/4 mile from the end of the runway. At this time power is decreased and flare started by pulling the nose up. The speed brake is extended, if necessary, to increase the drag. Forward visibility during the final approach is unacceptable. Poor forward visibility plus high stabilizer breakout forces and low airplane response to stabilizer movement cause the pilot, during the latter part of the approach, to more or less fly blind and "hunt" for the runway. During one night landing the landing was 2000 feet short of the end of the runway because the pilot was unable to see the runway during the final approach. In the approach configuration the airplane exhibits a yawing tendency and feels quite sluggish. The pilot has a tendency to keep his airspeed high and is definitely reluctant to reduce power to slow the airplane to the normal touchdown speed of 150 knots. It is essential that the low speed handling characteristics be improved prior to accepting the first production airplane and that a study be initiated immediately to increase the visibility over the nose. In view of the trend toward low

aspect ratio and the resulting higher angles of attack for landing, the requirement for good visibility during the approach and landing conditions may make it advisable to eliminate from consideration the use of nose inlets so that the cockpit may be placed in the extreme forward part of the fuselage to enable the pilot to have good visibility over the nose. Speed brakes were used on some of the landings but the high rate of sink with brakes retracted does not require their use. When speed brakes are utilized in the full down position the rate of sink is so high that it is necessary to add power. This is accompanied by a slight buffet which is objectionable because it feels as if the airplane is nearing a stalled condition. It would be desirable to incorporate a speed brake position indicator so that partial speed brakes could be used during the approach and landing flare. Immediately after touchdown speed brakes are extended to full open position and if the drag chute is not utilized, the nose wheel is allowed to contact the runway and immediate braking is applied. If the drag chute is used, the speed brake is extended immediately after touchdown and the drag chute employed shortly thereafter. When the drag chute opens there is a strong yawing tendency and it is possible that the airplane, if not controlled, could yaw off the runway to one side or the other. The best method of controlling this yaw is by use of nose wheel steering, inasmuch as airplane response to rudder is slow and no immediate corrective action is noticed. This yawing tendency during drag chute operation is unacceptable and should be corrected prior to delivery of the first production airplane. Because of the high landing speed of this airplane the drag chute should be used for all landings. Braking action is good but after the brakes heat up they become noticeably weak during the latter part of the landing roll. The wheel brakes need to be improved to insure full braking action throughout the entire landing roll. Use of the landing chute during all landings will decrease the wear on both the brakes and tires. The drag chute effectiveness is excellent but the time delay from actuation of the control handle to chute deployment is too long. Landings were made with cross winds up to 20 knots at 90 degrees to the landing path and they were executed by dropping the wing and crabbing into the wind. At touchdown, the wings were leveled and the airplane kicked straight, thus aligning the airplane with the runway immediately before or at touchdown.

8. High Altitude Operation:

The airplane performance is satisfactory at altitudes up to 40,000 feet. Maximum level flight speed at 35,000 feet is approximately 1.04 Mach number. Above 40,000 feet there is a 1 "g" buffet condition that is unexplained; it is possible that this may result from partial extension of the slats. On almost every flight above 40,000 feet, outboard portions of the slats were observed by the pilot to be extended from 1" to 2". If they were fully closed, it is believed that both the performance and the maneuverability of the airplane would be improved. The highest altitude attained during the Phase II tests was 55,000 feet. The airplane at this altitude, of course, is unusable as it barely flies and cannot be maneuvered at all without loss of altitude. It is believed that the highest practical altitude at which the airplane could be used would be approximately 50,000 feet. As previously mentioned, cabin pressure differential is excellent and the pilot's cockpit is comfortable. The cockpit heating system is adequate and the de-fogging system is superior to that of any other aircraft flown. The 2.75 PSI differential pressure provided in the YF-100A aircraft was never used and it should be eliminated from the production airplanes because it is doubtful if it will ever be used in tactical organizations.

9. Night Operation:

The YF-100A airplane was flown at night on two occasions during the Phase II tests. The first flight was aborted immediately after take-off due to the failure of the gear to retract. At this heavy gross weight, speed was increased during the final approach but because of the poor visibility over the nose and the inability of the landing light to light up the runway far enough ahead of the pilot to be able to see, touchdown was 2000 feet short of the runway at an indicated speed of 170 knots. During taxi and take-off the landing light, utilized as a taxi light, is satisfactory. During take-off roll the technique is no different than during daylight hours, except that, when the nose wheel is lifted prior to breaking ground, the pilot practically is on instruments inasmuch as he is unable to see over the nose to judge his attitude. Reference was made to the altimeter, and, upon reaching 200 feet, the airplane was leveled off and speed was increased to recommended climb speed. The cockpit lighting is excellent; however, the right instrument panel lights reflect on the left forward canopy and the left forward flood light reflects on the right forward canopy. This should be easily remedied and the night lighting, as a whole, is considered superior to that in any other fighter flown. Inasmuch as the landing light has a separate switch for extension or retraction, it is recommended that a safety device be installed to automatically retract and turn off the landing light when the gear is retracted to prevent inadvertent failure of the pilot to operate the switch.

10. Engine Operation:

The engine operation during the Phase II tests was satisfactory except for two major deficiencies that should be corrected prior to delivery of the first production airplane. One deficiency is the engine surge bleed valves. These bleed valves shut at approximately 90 percent rpm, thus increasing the thrust appreciably. This would make it quite difficult to fly formation at these power settings. A study should be made to eliminate sudden thrust increases when the bleed valves shut. The other deficiency is the engine power control. The present engine power control is hydro-electrical and gives an extremely non-linear engine response to throttle movement. This results in a slow, sluggish engine response to throttle control application. Further, there is a compressor stall which occurs when accelerating the engine from low power settings at approximately 70 percent rpm. This is undesirable as, upon one occasion, a compressor stall resulted when adding power during the final approach.

11. General Aircraft and Systems Functions:

a. Emergency systems.

The emergency gear and speed brake system, although not actuated, appears to be satisfactory. The standard emergency flight control system, which is actually one-half of the normal system, also appears to be satisfactory inasmuch as there is little chance that both rotors in J-57 type engines would fail at one time. Further, in case of engine failure, it is very doubtful that, due to the weight and touchdown speeds of this aircraft, dead stick landings will be attempted except under the most ideal conditions. The additional air driven turbine provided as a third emergency hydraulic source in the YF-100A airplane will not be installed in the production airplanes. A few power-off landings were made on the dry lake surface, but it was quite difficult to judge the touchdown point, and the pilot would be reluctant to attempt a dead stick landing on a runway unless he had considerably more experience

in the airplane and a fairly long runway on which to make the attempt.

B. PERFORMANCE

1. Take-Offs:

Performance take-offs to determine the minimum distance required to take-off and climb to 50 feet were made at 25,000 lbs. take-off weight and a mid cg position of 31 percent MAC. Minimum distance take-offs were accomplished by using full throttle with afterburner, contractor's recommended trim setting, and full nose up stabilizer at the point of take-off. Several take-offs were made by applying full nose up stabilizer prior to the point at which take-off speed was reached. Due to the increased drag on the aircraft with the stabilizer deflected in the full nose up position the take-off distance was appreciably increased when this technique was used. Take-offs with afterburner off are not recommended because of the excessive distance required. On one occasion, without afterburner and on a 30°C day at 2300 feet altitude, the airplane went off the end of an 8,000 foot runway before becoming airborne. The following table is a summary of the average take-off data corrected to NACA standard day, no wind, sea level conditions.

TAKE-OFF PERFORMANCE

Gross Weight lbs	Ave. Ground Run feet	Ave. Total Distance to 50'	Ave. IAS at T.O. knots	Ave. IAS at 50' knots	S.L. V_t at T.O. knots	S.L. V_t at 50' knots	Ave. Gross Thrust at T.O.-lbs	Afterburner Operation
25,000*	2990	4390	146	-	142	165	11,500	on
25,000**	5270	--	147	-	143	-	7,700	off

* average of six take-offs
** one take-off only

2. Climbs:

Climbs were flown using maximum and military rated power for gross weight of 24,300 pounds at sea level and best climb speed. The oil cooler was set on automatic and the gap varied from 0° at low altitudes to 18.4° at 54,000 feet. Climb speeds used were those recommended by the contractor for optimum rate of climb. These data have been corrected to standard day atmospheric conditions and are presented in Figures 2 and 3. A summary of these data is presented in the following tables:

MAXIMUM POWER CLIMB PERFORMANCE
Afterburner On

Altitude feet	R/C ft/min	T/C min	N_2 rpm	Fuel used - lbs	Naut. air miles travelled	Gross Wt--lbs	True Airspeed knots
S. L.	14,800	0*	9,510	0**	0	24,300	510
10,000	14,500	.7	9,510	300	6	24,000	520
20,000	13,900	1.4	9,510	575	12	23,725	535
30,000	10,700	2.2	9,350	825	20	23,475	540
40,000	6,600	3.35	9,195	1070	30	23,230	525
50,000	2,000	5.9	9,195	1380	52	22,920	520
53,800(SC)	100	10.6	9,195	1850	100	22,450	510

* Add 2.0 minutes to get time from brake release. (approx.)
** Add 800 lbs of fuel to account for start, taxi, take-off and accelerate to best climb speed. (approx.)

MILITARY POWER CLIMB PERFORMANCE
Afterburner Off

Altitude feet	R/C ft / min	T/C min	N_2 rpm	Fuel used-- lbs	Naut. air miles travelled	Gross Wt --lbs	True Airspeed knots
S. L.	4,300	0*	9,510	0**	0	24,650	425
10,000	4,000	2.3	9,510	300	19	24,350	455
20,000	3,500	5	9,410	550	40	24,100	475
30,000	2,250	8.4	9,180	800	67	23,850	492
40,000	650	15.9	9,070	1170	130	23,480	500
43,400(sc)	100	26	9,100	1525	220	23,075	500

*Add 3 minutes to get time from brake release. (approx.)
**Add 700 lbs of fuel to account for start, taxi, take-off and accelerate to best climb speed. (approx.)

3. Level Flight:

a. Level flight performance data were obtained at various altitudes for gross weights that would be obtained on check climbs at each respective altitude. Comparative performance data with two empty 275 gallon external fuel tanks attached were also obtained. Oil cooler operation was automatic and the gap varied from 3° to 18.5° open. All data have been corrected to an NACA standard day at the gross weights and altitudes presented in the following tables. True airspeed versus altitude data are presented in Figure 1 and are summarized in the following table:

TRUE SPEED -- KNOTS

TRUE SPEED AND MACH NUMBER

Altitude feet	Gross Wt. lbs	Maximum Power V_t knots	Maximum Power Mach No.	Maximum Power N_2 rpm	Military Power V_t knots	Military Power Mach No.	Military Power N_2 rpm
S. L.*	24,300	634	.96	9,510	542	.82	9,510
1,500	24,200	631	.96	9,510	552	.84	9,510
11,500	24,000	615	.97	9,510	589.5	.93	9,510
27,000	23,550	609	1.02	9,508	580.5	.973	9,420
35,000	23,350	600	1.04	9,345	562	.977	9,185
41,500	23,150	586	1.015	9,305	533	.929	9,125
44,000	23,100	582	1.005	9,295	520	.905	9,115
49,500	22,900	562	.98	9,260	—	----	-----

*Values extrapolated to sea level.

b. Curves of nautical air miles per pound of fuel versus Mach number are presented in Figures 4 and 5. The results of these data indicate that the best range of the aircraft can be realized by flying at as high an altitude as practicable. These results are summarized in the following table for recommended cruise settings. JP-4 fuel was used for all tests.

RECOMMENDED CRUISE PERFORMANCE

Altitude feet	Gross Wt. lbs	Mach No	CAS knots	TAS knots	N_2 rpm	Nautical air mi. per lb. of fuel
11,500	24,500	.675	366	428	8,790	.14
27,000	23,550	.78	315	465	8,620	.22
27,000 *	23,550	.775	313	462	8,790	.188
41,500	23,150	.90	265	517	9,030	.253

*External tanks attached.

c. Power required data were obtained from all level flight test points and are presented in Figures 6 through 9. It is of interest to note that between 234 and 322 knots CAS less rpm is required to maintain level flight at 27,000 feet than is required at 11,500 feet. This same trend of less rpm at higher altitude for the same CAS was experienced in the XB-52 aircraft which is also powered by J-57 engines. The results of these tests are summarized in the following table:

POWER REQUIRED AT ALTITUDE

				Calibrated airspeed -- knots					
Altitude feet	Gross Wt. lbs	Max rpm N_2	Mil rpm N_2	Max rpm	Mil rpm	9200 rpm	9000 rpm	8800 rpm	8600 rpm
11,500	24,000	9,510	9,510	535	512	464	424	368	287
27,000	23,550	9,508	9,420	425	402	385	364	341	310
27,000*	23,550	9,490	9,400	405	389	369	347	314	-
41,500	23,150	9,305	9,125	306	276	-	257	-	-

*External tanks attached.

d. Drag data for non-afterburning operation were obtained by use of calibrated tail pipe pressure probes located aft of the turbine and forward of the afterburner. No drag data were obtained for afterburner operation. These data are presented in Figures 10 through 15 and are summarized in the following table:

AIRCRAFT DRAG SUMMARY

	Clean Aircraft	2 Empty 275 gal. tanks
Minimum drag--lbs.	1,690	2,080
Equivalent airspeed for Minimum drag -- knots	250	265
f--sq. feet at min. drag	8.67	---
C_{D_p}(at minimum C_L)	.012	--

4. Airspeed Calibrations:

No standard production airspeed system was installed in the test aircraft. A test nose boom in which both the total and static pressure sources were located, was attached to the nose section of the aircraft above the air intake as shown in Figure 25. Angle of sideslip and angle of attack vanes were also attached to the nose boom aft of the static source. The system was calibrated by means of tower fly-bys at 2300 feet altitude and airplane fly-bys past the AFFTC F-86 pacer aircraft, USAF No. 48-209 at 40,000 feet. Two pace points were also obtained by using the pacer aircraft. One point was obtained from the ground radar for a Mach number of 1.39 in a dive. These data are presented in Figure 25 as a plot of ΔV versus indicated mach number.

5. Landings:

Performance landings to determine the minimum runway distance required to land over a 50 foot obstacle were made at an average gross weight of 22,000 pounds. No actual test data were obtained over a 50 foot obstacle due

to the extremely low approach made on the test landings. The aircraft was below 50 feet before its height could be recorded. Maximum performance landings were made by extending the speed brakes and chute immediately after touchdown and then using maximum braking to stop. Normal performance landings were made utilizing the same procedure as outlined above except the drag chute was not used. The landing data have been corrected to standard day, sea level, no wind conditions and the results are presented as follows:

LANDING SUMMARY

Ave. Gr. Wt--lbs	RPM N_2	Chute Deployed	IAS T.D.	S.L. V_t at T.D.-knots	Ground Roll - ft.
22,000	idle	yes	140	131	2880
22,000	idle	no	140	131	5440

6. Engine Static Thrust Calibration:

a. The engine used during the test program was operated on the AFFTC thrust stand to calibrate the exhaust gas total pressure probes and to determine the static performance of the XJ57-P-7 engine. These data are presented in Figures 16 through 22 and are summarized in the following table for standard day conditions at sea level:

STATIC ENGINE PERFORMANCE

N_2 rpm	N_1 rpm	Gross Thrust lbs	Exhaust Gas. - Temp °C	Specific Fuel Consumption lbs/hr -- lb	Surge Bleed Valve Position
9,495*	5,750	10,200	565	2.2	Closed
9,495 **	5,750	6,800	565	.91	Closed
9,400	5,580	6,250	540	.91	Closed
9,200	5,300	5,250	488	.915	Closed
9,000	5,030	4,480	440	.92	Closed
8,500	4,300	2,630	397	.975	Open
8,000	3,630	1,800	320	.98	Closed
7,500	3,100	1,300	270	1.02	Closed
7,000	2,720	980	258	1.14	Closed
6,500	2,430	730	255	1.39	Closed
6,000	2,160	520	255	1.73	Closed
5,640***	1,960	420	255	2.01	Closed

*A/B on - Max. power
** A/B off - Mil. power
*** Idle rpm

b. Reference to the above table indicates a considerable disagreement with the Pratt and Whitney model specification (maximum thrust of 13,200 pounds and military thrust of 8,450 pounds). A large percentage of this loss is attributed to the extremely low efficiency of the sharp lipped air intake on the YF-100A while the aircraft is not in motion. When the aircraft is airborne the ram efficiency reaches approximately 96 percent at 450 knots CAS.

C. STABILITY AND CONTROL

1. Test Configuration:

a. The configurations used throughout the test program and referred to in this report are based upon those set down in USAF Specifications 1815-B and conform to the following table. RPM, altitude, gross weight and trim speeds that were used for each test are listed on the individual curves.

TEST CONFIGURATIONS

Configurations	N_2 rpm	Trim airspeed	Landing Gear
Take-off	max.	-----	Down
Power*	max.	level flight	Up
Cruise	level flight	as noted	Up
Power approach	level flight	$1.4V_{s1}$**	Down
Landing	idle	$1.4V_{s1}$	Down

* All curves note whether afterburner is on or off.

** 1815 B states that the trim speed for the power approach configuration shall be $1.2V_{s1}$ but since the landing speed of the YF-100A is greater than $1.2V_{s1}$, all tests in the power approach configuration were conducted with the airplane trimmed at approx. $1.4\ V_{s1}$.

b. All Phase II tests were conducted at one cg range. No control over cg location was exercised while in flight as the fuel selection system was entirely automatic. The cg at engine start was 31.9 percent MAC and this value varied as fuel was consumed to a most forward cg of 29.5 percent MAC and to an aft cg of 31.7 percent MAC.

2. Mechanical Characteristics of the Control System:

a. The flight control hydraulic system of the YF-100A aircraft powers both the stabilizer and the ailerons. Two completely independent systems, each of equal output are provided and operate in parallel. Should

either system fail the other has sufficient capacity to power the controls. The rudder operates from the regular utility hydraulic system. Should the utility system fail, the rudder may be operated manually since the air loads on the rudder may be overcome by normal pilot effort. In the event of engine failure the windmilling engine RPM is sufficient to provide adequate hydraulic pressure at airplane speeds down through landing. In the event the engine "seized" (both rotors locked) an emergency air turbine pump is provided which operates by ram air. This ram air turbine will not be included in the production airplanes.

b. Longitudinal control is obtained through movement of the horizontal stabilizer, hydraulically actuated by means of two individual sets of cables and push-pull rods connected to the stick. The stabilizer is operated by an irreversible hydraulic power control and artificial feel is provided by means of an artificial feel bungee. Forces in maneuvering flight are augmented by means of a two pound per "g" bobweight. The bobweight force varies from a maximum of 2.5 pounds per "g" at zero degrees attitude with full nose down stabilizer to a minimum of .12 pounds per "g" at 20° attitude with full nose up stabilizer. The two pounds per "g" figure quoted above is for the aircraft trimmed for cruise conditions at zero degrees attitude. A schematic drawing of the longitudinal control system is presented on page 10, Appendix II. There is no airspeed sensing device ("q" box) incorporated in the system; therefore, control forces are always the same for stick deflection from trim position in static flight. Longitudinal trim is obtained by means of moving the stabilizer through an electrical trim motor.

c. Lateral control is obtained by irreversible hydraulically operated ailerons actuated by push-pull rods and cables connected to the stick. The aileron control system consists of two mid-span ailerons on the trailing edge of each wing. Artificial feel is provided, as with the stabilizer, by an artificial feel bungee. During the Phase II tests, "in-flight" deflection of the ailerons was restricted to two-thirds stick throw above .6 Mach number due to the lack of structural integrity tests; however, some data were obtained at approximately 80 percent stick throw above .6 Mach number with no adverse results. No static balance is incorporated in the lateral control system. Lateral trim is maintained through an electric trim actuator which moves both the inboard and outboard ailerons on each wing by means of the hydraulic power unit. A schematic diagram of this system is found on page 11, Appendix II.

d. Directional control is maintained in a manner similar to the longitudinal and lateral systems. An irreversible hydraulic actuating cylinder moves the rudder and is controlled by cables and push-pull rods connected to the rudder pedals. Artificial feel is provided by a bungee and directional trim is maintained by an electric trim actuator which moves the rudder by means of actuating the hydraulic cylinder. A schematic drawing of the directional control system is illustrated on page 12, Appendix II.

e. During all Phase II tests it was found that all trim devices would reduce the control forces to zero and maintain settings.

f. Control friction was measured in a closed hangar at a temperature of 20 °C. Breakout forces were measured as that force required to start the control surface to move at any point over the entire range of a control system.

Both the rudder and the stabilizer break-out forces are excessive and above the required values of Specification 1815B. The aileron break-out forces are within the limits of 1815B and are satisfactory. Break-out forces in flight were checked on several occasions and though the in-flight forces are somewhat less than the static break-out force, the stabilizer and rudder break-out forces are still felt to be excessive. Plots of force versus control surface deflections are presented in Figures 28, 29 and 30 for the longitudinal, lateral and directional control systems respectively. The results of these tests are briefly summarized in the following table:

CONTROL FRICTION
Ground Test Data

20 °C Air Temperature

Control	Break-Out Force at Trim--lbs	Ave. Break-Out Force Over Control Range -- lbs	Allowable Limit 1815B--lbs
Stabilizer	8	10	4.5
Aileron	2	3	3
Rudder	22	15	10.5

3. Stabilizer Control Effectiveness:

a. Take-off tests were conducted from a concrete runway and from Rogers Dry Lake bed to determine the ability of the stabilizer to produce take-off attitude. No quantitative data were obtained but, using the contractor's recommended trim setting of 4° nose up and full nose up stabilizer at take-off, control is more than adequate at a weight of 25,000 pounds and a cg of 31 percent MAC. With the large amount of stabilizer control which is available, it is believed that stabilizer control effectiveness will be satisfactory for all conditions of loading and cg. Stick forces at the point of nose gear lift-off are light and estimated to be under the maximum of 35 pounds set forth in USAF Specification 1815B.

b. Landing tests were conducted to determine the ability of the longitudinal control to hold the aircraft off the runway until the maximum allowable ground angle was reached. Qualitative tests were conducted at gross weights of 21,000 to 22,000 pounds and approximately 30 percent MAC. Quantitative data recorded on Figure 31 for one landing indicates that the tail can be dragged at speeds of 136 knots or below at a cg of 30 percent MAC. As during the take-offs, adequate stabilizer is available at all times to produce the maximum ground angle. Care had to be exercised on all landings so as to insure that the tail was not dragged inadvertently prior to touchdown as was done on the above mentioned landing. It is estimated that sufficient control would be available for all practicable landing cg positions and gross weights. Though the control break-out forces are excessive and aircraft response to control movement is slow and sluggish on landings, the forces are light and are under the 1815B limit of 35 pounds pull force.

4. Stalls:

a. The stall characteristics are considered generally unsatisfactory. In the power approach configuration at 10,000 feet and power configuration at 35,000 feet with the speed brake out, there is an uncontrollable left roll-off at minimum airspeed. In the power configuration at 35,000 feet and 45,000 feet there is a slight pitching tendency 20 to 30 knots above the minimum flying speed and a rolling and yawing moment from about 1.25 V_s down to the minimum flying speed. The pitch-up is not violent and is easily controlled. The banking and yawing may be controlled by judicious use of the ailerons and rudders until the aircraft reaches the minimum flying speed, at which point, with the stabilizer full nose up, the aircraft "mushes" through several thousand feet of altitude with no apparent tendency to roll-off on either wing. Control is adequate during the approach to, and recovery from, the stall. Since the stall occurs 30 to 40 knots below the landing speed of the aircraft, it is felt that the poor stall characteristics are tolerable but should be remedied as soon as possible. No other unsatisfactory characteristics are apparent at high altitude or under accelerated conditions.

b. Time histories of the power approach configuration stalls and the power configuration stall at 45,000 feet are presented in Figures 32 through 34 and are summarized in the following table:

STALL CHARACTERISTICS

Configuration	Altitude feet	Minimum IAS - knots	Type of and IAS for First Warning
Power 21,500 lbs 30 percent MAC	45,000	115	Slight pitch-up -141 knots
Power Approach 21,200 lbs 30.5 percent MAC	8,000	105	Yawing tendency --131 knots

5. Static Longitudinal Stability:

a. Static longitudinal stability tests were conducted at 15,000 and 35,000 feet in the power condiguration and 10,000 feet in the power approach configuration at an average cg of 31 percent MAC and an average gross weight of approximately 22,000 pounds. No neutral points were obtained since the tests were flown at one cg range only. Pilot technique involved trimming the aircraft at the desired airspeed and altitude and then varying the airspeed by use of the stabilizer alone without changing either the trim or throttle setting.

b. In the power configuration at both 15,000 and 35,000 feet, the static longitudinal characteristics are unsatifactory. At 35,000 feet between .80 and .97 Mach number and at 15,000 feet between .86 and .99 Mach number

the aircraft exhibits either neutral or negative stability. A pull force of 7.5 pounds is required to maintain a trim speed of 323.5 knots at 35,000 feet when the afterburner is ignited, indicating a slight nose down pitching moment. However, this is not too objectionable as the forces are easily trimmed out. This trim change is graphically presented in Figure 35. Data obtained in the power approach configuration indicate no unsatisfactory characteristics. It is possible that, if a steeper force gradient could be obtained, the slow and sluggish low speed handling characteristics may be improved somewhat. Plots of stick force and stabilizer position versus calibrated airspeed are presented in Figures 35 through 37.

6. Dynamic Longitudinal Stability:

a. The dynamic longitudinal stability was investigated at 45,000 and 10,000 feet in the power configuration and at 10,000 feet in the power approach configuration at an average of 31 percent MAC. The testing technique involved making abrupt stabilizer deflection from trimmed level flight in order to obtain a positive acceleration of approximately 2.0 "g's" and a negative acceleration of approximately 0.0 "g". Tests were conducted with the controls free and with the pilot attempting to reduce the oscillations as rapidly as possible. No control fixed tests were made because, with the irreversible control system, the result would be the same as with controls free.

b. Test results indicate that at 45,000 feet the aircraft fails to satisfy the requirement of Specification 1815B (the short period normal acceleration oscillation shall be damped to 1/10 amplitude in one cycle). With controls free the oscillation persists for as long as 7 to 8 seconds and 4 to 5 cycles. When an attempt is made to stop the oscillations, pilot technique is the determining factor as to the number of cycles required to damp out the normal acceleration oscillations. In one case the oscillations were damped to 1/10 amplitude in two cycles by the pilot's judicious use of the controls. At 10,000 feet and 475 knots CAS, the dynamic longitudinal stability with controls free is satisfactory. When controls are used to damp the oscillations there may be a "g" overshoot, which in one case, was greater than the initial acceleration. If care is not exercised during a control displacement of this type, it is possible to cause a pilot induced oscillation similar to that experienced in the early F-86D model aircraft. In the power approach configuration the dynamic longitudinal stability is satisfactory. The normal acceleration oscillation damps in from one-half to one cycle. Under no conditions are there any undamped short period stabilizer oscillations.

c. These data appear as time histories in Figures 28 through 47 and are summarized in the following table:

DYNAMIC LONGITUDINAL STABILITY

Configuration	CG %MAC	CAS knots	Alt. ft	Cycles to damp to 1/10 amplitude	Sec. to damp to 1/10 amplitude
	CONTROLS FREE				
Power(A/B on)	31	274	45,300	5	8
Power(A/B on)	31	274	45,300	4+	7+
Power(A/B off)	31.2	475	10,500	1.5	3.5
Power(A/B off)	31.2	475	10,500	2	3.5
Power approach	30.1	164	9,900	1/2	3.5
Power approach	30.1	164	9,900	1	4.5
Pilot Attempting to Reduce Oscillations as Rapidly as Possible					
Power(A/B on)	31	274	45,300	5	7.5
Power(A/B on)	31	274	45,300	2	3
Power(A/B off)	31.2	475	10,500	1.5	2.5
Power(A/B off)	31.2	475	10,500	2	3.5

7. Maneuvering Flight Characteristics:

a. Maneuvering flight characteristics were tested in the power and cruise configurations at 46,000 feet and at 12,000 feet in the power and power approach configurations. Tests were conducted by placing the aircraft in steady turns to obtain stabilized conditions of acceleration, stick force and speed while allowing the altitude to vary as little as possible.

b. Although the design limit load factor for this aircraft is 7.33 "g's", the contractor imposed a temporary restriction of 5.0 "g's" for all Phase II tests. With the exception of the power approach configuration the maneuvering characteristics are generally satisfactory in the range between stick force break-out and force lightening or pitch-up. The stabilizer position gradients are positive, up to pitch-up or force lightening, for all test conditions except the power approach configuration, thus indicating the stick fixed neutral points to be well aft of the cg at which the tests were conducted. After the initial break-out force was applied the forces were, in all cases, proportional to stabilizer deflection. At 45,000 feet, with the aircraft trimmed at 282 and 271 knots CAS, the stick force gradient is slightly higher than the 1815-E maximum of 8.85 pounds per "g". This is not excessive and it is felt that this condition is tolerable. For all other conditions in the power and cruise configurations the force gradient is less than the Specification maximum limit. In the power approach configuration the maneuvering characteristics are unsatisfactory. The stick fixed and stick free neutral points for the power approach configuration appear to be at approximately the cg at which the tests were conducted (31 percent MAC). The true force picture, in the approach configuration, was masked by the high break-out forces but the data indicate a force gradient approaching zero pounds per "g". These data are presented in Figures 48 through 53 and are

summarized in the following table:

MANEUVERING FLIGHT CHARACTERISTICS

Configuration	CG	Ave. Force gradient	Max. "g" attainable	Max. 1815-B gradient	Min. 1815-B gradient
	% MAC	lbs/"g"	"g"	lbs/"g"	lbs/"g"
Power (A/B on) 282 knots CAS 45,800 feet 23,400 lbs	30.5	10.5	1.8	8.85	3.32
Power(A/B on) 271 knots CAS 47,000 feet 22,000 lbs	29.9	10.0	1.85	8.85	3.32
Cruise 254 knots CAS 45,900 feet 21,670 lbs	29.8	8.0	2.0	8.85	3.32
Cruise 237 knots CAS 45,700 feet 21,300 lbs	30.3	5.5	1.75	8.85	3.32
Power (A/B off) 484 knots 11,950 feet 23,000 lbs	31.7	4.3	5.0*	8.85	3.32
Power Approach 159 knots IAS 11,190 feet 21,900 lbs	31	0	1.5	No requirement	

*Flight test limit load factor

c. Additional data were obtained in the power configuration at 45,000 and 10,000 feet to demonstrate that the stick force gradient during quick pull-ups exceeded that obtained in steady turns. These data are presented in Figures 48 and 52, and as time histories in Figures 54 and 55, and show that Specification 1815-B requirements are satisfied.

d. Data were also obtained to determine the maximum "g" available in supersonic dives to approximately 1.07 Mach number at 35,000 feet. This test was accomplished by diving the aircraft from 40,000 feet and applying back stick in a diving turn at approximately 35,000 feet. The time histories of these tests, presented in Figures 56 and 57 indicate that a maximum of 6 "g's" are available at 349 knots IAS 1.0 Mach number, at 35,000 feet. However, an acceleration of this magnitude and speed cannot be held as there is a very rapid bleed-off in airspeed.

8. High Speed Flight Characteristics:

The high speed flight characteristics were investigated in dives from altitudes above 50,000 feet. The flying qualities of the YF-100A aircraft are excellent throughout the high speed range up to a Mach number of 1.39, the highest speed obtained. A time history of one dive to this Mach number is presented in Figures 58 and 59. A slight trim change, as shown by the change in stabilizer position, aileron position and rudder-force, was encountered at approximately 1.15 Mach number. This is not considered unsatisfactory as it was not noticed by the pilot at the time of the test. Data obtained from the AFFTC Radar-Phototheodolite are also plotted in Figure 58. These data show that the indicated Mach number of 1.39, which was obtained in this dive, was the same as the true Mach number determined from the true airspeed obtained from the radar data.

9. Speed Brake Operation:

a. Time histories of speed brake operations are presented in Figures 60 and 61. From trimmed, level flight at a calibrated airspeed of 590 knots at 3,500 feet, the speed drops to 370 knots in 200 seconds when the speed brake is open and the altitude held constant. At this speed, the time required for the brake to open fully is 13 seconds. In this period, the airspeed drops to 520 knots. The initial movement of the speed brake is accompanied by a nose down trim change. In the test shown in Figure 60, this trim change was at least 0.5 "g" but was easily controlled with less than 15 pounds of force. It is noted that, in this instance, the attempt to compensate for the trim change resulted in a longitudinal oscillation, because of the poor dynamic stability characteristics of the airplane, which produced a normal acceleration of 2.5 "g" even though the pilot was attempting to maintain level flight.

b. In the power approach configuration, the speed brake was opened at a stabilized speed of 160 knots at 4,300 feet. The airspeed was held constant and the resulting altitude loss was only 700 ft/min after the brake was extended. At this speed the speed brake extends fully in approximately 2.5 seconds. A slight nose down trim change occurs only after the speed brake reaches a half open position. The pull force required to compensate for the trim change is less than 10 pounds.

10. Static Directional Stability:

a. The static directional stability was investigated at 35,000 feet in the power configuration and at 10,000 feet in the power approach configuration. The airplane was placed in steady sideslips by deflecting the rudder while using sufficient lateral control and bank angle to hold a constant heading.

b. The sideslip characteristics are satisfactory. With the exception of the high rudder break-out forces (18 to 24 pounds), the sideslip angle was substantially proportional to the rudder pedal force and rudder deflection. In the power configuration 5.8 degrees of sideslip is attained at 57 pounds of rudder force. In the power approach configuration 11 degrees of left sideslip is attainable before full aileron is required to hold a constant heading.

c. The results of these tests appear in Figures 62 and 63 and briefly in the following table:

STATIC DIRECTIONAL STABILITY

Configuration CAS Alt Wt.	Stabilizer force at 50 lbs Rud. Force - lbs		Maximum sideslip angle degrees		Maximum rudder deflection degrees		Rudder force at Maximum deflection-lbs		L. outer aileron deflection at max. rudder degrees	
	RUDDER		SIDESLIP		RUDDER		RUDDER		RUDDER	
	Lt.	Rt.	Lt.	Rt.	Lt.	Rt.	Lt.	Rt.	Lt.	Rt.
Power 281 knots 45,970 feet 21,700 lbs	5	--	--	5.7	18.5	--	57	--	--	2
Power approach 160 knots 11,620 feet 23,500 lbs	3	--	13	12	18	14	60	45	15	15

11. Dynamic Lateral Directional Stability:

a. Dynamic lateral directional stability tests were flown to determine the response of the airplane to either lateral or directional disturbances and to determine if there was any tendency for any oscillation of the rudder or aileron control surfaces. Tests were made at 45,000 feet and 10,000 feet in the power configuration and at 10,000 feet in the power approach configuration. Only control free tests were made as it was felt that results of control fixed tests would be the same because of the irreversible control system. Tests were first made by kicking the rudder abruptly to induce an oscillation. Oscillations were then induced by abruptly deflecting the ailerons. Both procedures were repeated again but with the pilot attempting to reduce the oscillations as rapidly as possible by using the controls.

b. Data obtained are presented in Figures 64 to 83. In all cases, except in the power approach configuration, the dynamic lateral directional stability is unsatisfactory. The time for the oscillations to damp to 1/2 amplitude is excessive and there is a tendency for small amplitude oscillations to persist for 10 to 12 seconds. There is no indication of control surface oscillations. In the power approach configurations, the time to damp to 1/2 amplitude is satisfactory and all oscillations appear to damp out in 8 to 10 seconds. The oscillations may be reduced more rapidly by use of the controls.

12. Adverse Yaw:

a. The sideslip developed during rudder fixed abrupt rolls from a steady banked turn was measured at 167 knots indicated airspeed at 10,000 feet. This was approximately sixty percent above the stalling speed for this configuration. With full aileron deflection (15°), the adverse yaw produced indicated values of sideslip of 19° to the left in left rolls and 11° to the right in right rolls. Rolls were then repeated using the rudder to keep the ball in the cockpit turn indicator centered. At this speed, 13° of rudder (20° available) is sufficient to keep the ball centered. Time history plots of these tests are presented in Figures 84 and 85. It should be noted that, when rolling with the ball in the center, the sideslip vane mounted on the nose boom gave indicated sideslip angles of as much as 6 to 8 degrees in the direction of the roll. It is apparent that, owing to the location of the vane and to the rotation of the airplane about its cg in a maneuver of this nature, at a high angle of attack, erroneous sideslip indications are obtained. Assuming that the error also exists in the rudder fixed abrupt aileron rolls, it is likely the actual sideslip developed as a result of the adverse yaw is only in the neighborhood of 8 to 10 degrees. Although these tests were run at a speed somewhat faster than the more critical speed of 1.4_{SL} required by the specification, it is felt that directional stability and rudder power are both adequate to satisfactorily overcome the effects of the adverse yaw.

b. Similar tests were made in the power configuration at 245 knots ($2.0\ V_s$) at 45,000 feet. Maximum aileron travel in these tests was limited to 13° because of a temporary restriction on the airplane. The data obtained are presented in Figures 86 and 87. Again it appears that the indicated values of sideslip are in error. At this speed only 5° of rudder is required to keep the ball in the cockpit turn indicator in the middle when using the rudder to offset the effects of the adverse yaw. The true sideslip resulting from the rudder fixed rolls is only 2 or 3 degrees.

13. Lateral Control:

a. Rate of roll tests were conducted at 45,000 feet in the power configurations (A/B on and off), at 10,000 feet in the power configurations (A/B on and off), and in the power approach configurations at 10,000 feet. These tests were repeated using rudder to assist the roll for all conditions except the power configuration with the afterburner on. All rolls were accomplished by starting from a 30 to 60 degree banked turn and then abruptly applying maximum allowable aileron deflection to roll out of the turn and continue through at least 360° (70 to 90 degrees in the power approach configuration). The rudder pedals were either held fixed at the trim position or rudder was applied

with the ailerons to coordinate the roll. As mentioned previously in the report, during all Phase II tests the aileron deflection was restricted to 2/3 stick throw above .6 Mach number because of the lack of structural integrity tests on the aircraft.

b. Because the pilot was unable to use full aileron, a direct comparison to the maximum requirement of Specification 1815-B for the power configuration tests cannot be made. By extrapolation of the data available, it is felt that the rolling characteristics are generally satisfactory. At 45,000 feet, at maximum level flight speed (278 knots CAS), roll rates of 190 degrees per second can be achieved at approximately 80 percent aileron deflection. The tests at 10,000 feet in the power configuration indicate that at high indicated airspeeds (above 500 knots) the aileron effectiveness is decreased somewhat. With the aircraft trimmed at 462 knots roll rates of 230 degrees/sec can be obtained at approximately 80 percent aileron deflection while at 538 knots roll rates of 160 degrees/sec are all that are obtainable at the same aileron deflection. The rudder has little or no effect on the roll rate at either 45,000 or 10,000 feet in the power configuration. With the aircraft trimmed at approximately $1.4V_{SL}$ in the power approach configuration the roll rate is inadequate. For right rolls the helix angle obtainable is .06 while for left rolls the helix angle is only .038 as compared to the 1815-B requirement of .07. The use of rudder increases the roll rate by 10 to 30 percent but is still under the specified helix angle requirement of .07. In all cases, both in the power and power approach configurations the aileron forces are well under the specification requirement of 30 pounds stick force.

c. The aileron roll data are presented in summary form in Figures 88 and 89 and as aileron characteristics in Figures 90 through 92. Time histories of the roll data are presented in Figures 93 and 94. The data are summarized in the following table:

AILERON ROLLS

Configuration	Dir. of Roll	Rudder Pos. deg.	L. Outer ail. pos. deg.	Aileron force lbs.	Helix angle Pb/2V radians	Roll Rate deg/sec
		RUDDER PEDALS FIXED				
Power(A/B on)	Left	2 rt.	11.5 up	12	.047	145
45,000 ft.	Right	2 rt.	12.5 dn	12.5	.063	192
278 knots CAS						
Power(A/B off)	Left	2 rt.	11.5 up	12	.051	140
45,000 ft	Right	2 rt.	13.5 dn	13	.066	175
245 knots CAS						
Power(A/B on)	Left	0	11.7 up	15	.043	145
10,500 ft	Right	0	10.3 dn	14	.04	134
538.5 knots CAS						

(AILERON ROLLS con't)

Configuration	Dir. of Roll	Rudder Pos. deg.	L. Outer ail. pos. deg.	Aileron force lbs.	Helix angle Pb/2V radians	Roll Rate deg/sec
Power (A/B off) 10,500 ft 462 knots CAS	Left	0	12.4 up	15	.08	233
	Right	0	11.5 dn	14	.076	218
Power Approach 10,400 ft 167 knots CAS	Left	2 rt.	14 up	--	.037	40
	Right	2 rt.	15 dn	--	.061	65
RUDDER USED TO ASSIST ROLL						
Power (A/B off) 45,000 ft 244 knots CAS	Left	3 lt.	1.2 up	--	.055	147
	Right	8 rt.	13.5 dn	--	.067	183
Power (A/B off) 10,500 ft 462 knots CAS	Left	4 lt.	12.4 up	--	.079	230
	Right	7 pt.	12.3 dn	--	.078	225
Power Approach 10,500 ft 167 knots CAS	Left	12 lt.	14 up	--	.0495	53
	Right	14 rt.	15 dn	--	.067	70

Note: Full static aileron deflection= 15° up and down

D. DATA ANALYSIS METHODS

1. Introduction:

This section briefly discusses the methods of data reduction that were used in analyzing the test data. The following references were used and will be referred to in the succeeding discussion.

No. 1 "Flight Test Engineering Manual", USAF, Technical Report No. 6273.

No. 2 "Performance Flight Testing Methods in Use by the Flight Section", USAF, Technical Report No. 5069.

No. 3 "Pressure Altitude Method of Flight Test Data Reduction", AMC Memorandum Report No. TSFTE 2060.

No. 4 "A Method of Determining Delta Rate of Climb for Turbo-Jet Powered Aircraft," AMC Memorandum Report No. MCRFT 2157.

No. 5 "Flying Qualities of Piloted Airplanes", USAF Specification 1815-B.

No. 6 "Performance Flight Testing Methods Jet Propelled Aircraft," USAF Technical Report No. 5239.

No. 7 "Specification No. A-1639-B, Model XJ57-P-7 Engine," Pratt and Whitney Aircraft.

No. 8 "Standardization of Take-Off Performance Measurement for Airplanes," AFFTC Technical Note R-12.

2. Notation:

The following notation will be used throughout the succeeding discussion of methods of data analysis.

Symbol	Description	Units
A/R	Aspect ratio	----
b	Wing span	feet
C_L	Lift coefficient	----
C_D	Drag coefficient	----
D	Airplane drag	pounds
dh/dt	Apparent rate of climb	ft/min
e	Airplane efficiency factor	----
F_{gt}	Test gross thrust	pounds
F_{gs}	Standard gross thrust	pounds
Fe	Air flow momentum or ram drag	pounds
F_n	Net thrust	pounds
F_g	Gross thrust	pounds
ΔF_n	Net thrust on a standard day at standard rpm minus net thrust on a test day	pounds
g	Acceleration due to gravity	ft/sec^2
Δh	Altimeter position error	feet
K	Temperature recovery factor	----
K_tA_j	Equivalent nozzle area	feet^2
L	Lift	pounds
M	Mach number	----
P_a	Ambient air pressure	in. Hg
P_{t2}	Compressor inlet total pressure	in. Hg

Symbol	Description	Units
P_{t7}	Total turbine impact pressure	in. Hg
q_c	Impact air pressure	in. Hg
R/C_{std}	Standard day rate of climb	ft/min
N_{2s}	Standard day high pressure compressor rpm	rpm
N_{2t}	Test day high pressure compressor rpm	rpm
R_{at}	$W_t\ (E_t + 50)/S_{at}$	pounds
S	Area of wing	feet^2
S_{gs}	Standard day sea level ground distance	feet
S_{gt}	Test day ground distance	feet
S_{as}	Standard day sea level air distance to 50 feet	feet
S_{at}	Test day air distance to 50 feet	feet
S_{ts}	Standard day, sea level total distance from start to 50 feet or from 50 feet to stop	feet
T_{as}	Standard day ambient air temperature	°C
T_{at}	Test day ambient air temperature	°C
T_{ic}	Indicated free air temperature	°C
T_{t2}	Compressor inlet total temperature	°C
t	Time from 50 feet to touchdown or from T. O. to 50 feet	seconds
V_{to}	Velocity at take-off	ft/sec
V_{50}	Velocity at 50 feet	ft/sec
V_{td}	Velocity at touchdown	ft/sec
V_w	Velocity of wind down runway	ft/sec
V_c	Calibrated airspeed	knots
V_e	Equivalent airspeed	knots
V_t	True airspeed	knots
ΔV	Airspeed position error	knots
W_s	Standard gross weight	pounds
W_t	Test gross weight	pounds
ΔW	Standard minus test gross weight	pounds
W_a	Airflow through jet engine	lbs/sec
δ	Pressure ratio($P_a/29.92$)	----
η_R	Ram or duct efficiency	percent
σ	Density ratio	----
θ	Temperature ratio(T_{at}/T_{as})	----

3. Take-Offs:

Take-offs were made to determine the minimum total distance required to clear a 50 foot obstacle at both maximum and military power. These data have been corrected to standard day, sea level, no-wind conditions at standard gross weight and power. These corrections were accomplished by use of the following relationships derived in Reference No. 8:

$$S_{gs} = S_{gt}\left[\frac{V_{to} + V_w}{V_{to}}\right]^2 \left(\frac{F_{gt}}{F_{gs}}\right)^{1.3} \left(\frac{W_{std}}{W_t}\right)^{2.3} \left(\frac{\sigma_t}{\sigma_s}\right)$$

$$S_{as} = \left(S_{at} - tV_w\right) \left(\frac{F_{gt}}{F_{gs}}\right)^{1.6} \left(\frac{W_{std}}{W_t}\right)^{2.3} \left(\frac{\sigma_t}{\sigma_s}\right)^{.7}$$

Since no tailpipe instrumentation was available for measurement of afterburner thrust, the net thrust augmentation ratio (F_{na}/F_n) was used to approximate the total thrust being delivered. This ratio was determined from data obtained on the static thrust run (Ref. 1, page 3-37). All maximum and military power test thrust data were corrected to standard day sea level conditions and plotted as thrust versus rpm to determine the average gross thrust at take-off for rated engine rpm. These data showed an average gross thrust of 11,500 pounds at 9,495 rpm at the point of take-off for maximum power operation. A gross thrust of 7,700 pounds at the same rpm was obtained for military power operations.

4. Climbs:

a. Climb tests were flown using military and maximum rated power. Prior to starting the Phase II tests the engine was set to produce an rpm which Pratt and Whitney had predetermined as being the rpm which would produce 8,450 pounds of static thrust at military power and 13,200 pounds of static thrust at maximum power in a ground test cell on a standard day at sea level. This (N_2) was 9,495 rpm. Since the fuel control unit of the XJ57-P-7 engine maintained a predetermined schedule of engine rpm (N_2) with respect to compressor inlet temperature, it was necessary to determine this schedule. These data appear in the form of a bias curve in Figure 26, N_2 being plotted versus compressor inlet temperature. All climb data have been corrected to standard day atmospheric conditions; a limit exhaust gas temperature of 570°C up to 25,000 feet, 610°C up to 40,000 feet and 640°C above 40,000 feet, or military rpm and the gross weight that would have been obtained on a standard day if the climb started at sea level and best climb speed.

b. These corrections were accomplished as follows:

(1) $$R/C_{std} = dh/dt \sqrt{T_{at}/T_{as}} + \frac{101.2\, V_t\, \Delta F_n}{W_t}\sqrt{\frac{T_{as}}{T_{at}}}$$

Thrust data for these corrections were obtained from Reference No. 7. The derivation of this method is outlined in Reference No. 4. Weight corrections were applied by use of the following equations which are presented in Reference No. 2.

$\Delta R/C_1 = dh/dt \times \dfrac{\Delta W}{Wt}$ = Rate of climb change for ΔW at constant power

$\Delta R/C_2 = \dfrac{19010 \quad \Delta W}{V_c \quad \sigma^{1/2} \quad e\, b^2}$ = Rate of climb change for induced drag change because of ΔW.

$$e = C_L^2 / (C_D - C_{DP}) \pi AR$$

(2) The exhaust gas temperature was corrected to the temperature corresponding to the standard day value of $T_{t7} \times 288/T_a$ in the case where this parameter was less than that for the limit exhaust gas temperature. For the case where the test ratio of $T_{t7} \times 288/T_a$ exceeded the maximum, it was necessary to correct the rpm. These corrections were accomplished in the following manner:

(a) A plot of $T_{t7} \times 288/T_a$ versus $N \sqrt{\dfrac{288}{T_a}}$ was constructed from the test data.

(3) For the case where $T_{t7} \times 288/T_a$ was less than the allowable limit, the ratio was increased or decreased to a value corresponding to the standard day value of $N_{2s} \times \sqrt{288/T_a}$.

(4) For the case where $T_{t7} \times 288/T_{at}$ exceeded the allowable limit, the parameter $N_2 \sqrt{288/T_a}$ was decreased to a value corresponding to the limit standard day value of $T_{t7}\ 288/T_{as}$. The rate of climb change resulting from the change in thrust for this rpm correction was applied using the thrust data from Reference No. 7 and the equation:

$$\Delta R/C = \frac{101.2\, V_t\ \Delta F_n}{Wt} \sqrt{T_{as} / T_{at}}$$

which has already been discussed.

5. Level Flight:

Level flight data were obtained by flying each successive test point at an altitude selected to give a constant ratio of gross weight to atmospheric pressure, to provide a minimum weight correction for the basic data. All weight pressure ratios were preselected to represent approximately the values which would be obtained on a standard day by climbing to the selected altitude from a sea level gross weight of 24, 300 pounds. The methods of reduction are outlined in Reference No. 1. The equations used were as follows:

$$N_{2s} = \left[N_{2t} + \frac{(\Delta N\sqrt{288/T_a})}{\Delta W/\delta} \quad \Delta (W/\delta) \right] \sqrt{T_{as}/T_{at}}$$

$\dfrac{\Delta (N\sqrt{288/T_a})}{\Delta W/\delta}$ = Weight correction factor obtained from test data plots similar to Figure 6 at test Mach number.

To determine military or maximum rated rpm in level flight it was necessary to estimate from the test data the maximum attainable Mach number under the desired conditions and then to determine the standard day compressor inlet temperature which in turn determined the standard day military rpm by use of the rpm bias curve presented in Figure 26.

6. Thrust and Drag:

Thrust required and drag data were obtained during the level flight calibrations. These data were obtained from the total pressure and temperature pick-ups installed in the tail pipe aft of the turbine and forward of the afterburner. Thrust required and drag data for afterburner operation were obtained from the augmentation ratios as described in Paragraph 3 of this discussion. The pressure probes were calibrated during the static thrust tests conducted on the Edwards Air Force Base thrust stand. The calibration data have been presented in Figures 16 through 23. The calibration data have been extrapolated to include all conditions of flight by use of the ideal gas flow equations which are derived in Reference No. 6 and presented briefly as follows:

Subsonic: $$F_g/P_a = 570.06\,(K_t A_j) \left[\left(\frac{P_{t7}}{P_a}\right)^{0.2481} - 1 \right] \text{ for } \frac{P_{t7}}{P_a} < 1.85$$

Supersonic: $$F_g/P_a = 70.727(K_t A_j) \left[1.26 \frac{P_{t7}}{P_a} - 1 \right] \text{ for } \frac{P_{t7}}{P_a} > 1.85$$

No airflow data were obtained. The Pratt and Whitney Division recommended method for determination of airflow was utilized. The curve used for this determination is presented in Figure 27, Appendix I. To obtain the net thrust of the engine installation, the ram drag was subtracted from the gross thrust.

$$F_n = F_g - F_e$$

$$F_e = 0.0524\, W_a\, V_t$$

7. Lift and Drag Coefficients:

The lift and drag coefficients for incompressible flow were calculated from the following equations:

$$C_L = \frac{295 \ W_t}{V_e^2 \ S}$$

$$C_D = \frac{295 \ F_n}{V_e^2 \ S}$$

8. Airspeed Calibration:

The test boom airspeed system with a North American Aviation Inc. manufactured head was used throughout all tests and was calibrated in the clean configuration only. Calibrations up to approximately .98 Mach number were made by use of the Air Force Flight Test Center F-86A pacer, USAF No. 48-209. Calibration points above .9 Mach number were obtained by making airplane and tower fly-bys and using the altimeter method of airspeed calibration. Supersonic data were obtained in a dive and calibrated against a radar computed plot of the dive. The data and a schematic diagram of the nose boom installation are presented in Figure 23. The airspeed calibration curve is presented as airspeed correction to be added (ΔV) versus indicated Mach number. It is realized that this is a rather unorthodox presentation but by plotting the correction in this manner the data approached a common correction curve for all altitudes, for Mach numbers above .88.

9. Altimeter Correction:

The altimeter corrections for position error were computed from the airspeed calibrations data by use of the following expression derived in Reference No. 1.

$$\frac{\Delta h}{\Delta V} = \frac{V_c \left[1 + \frac{V_c}{2,183,944} \right]^{2.5}}{11,282 \ \sigma}$$

10. Temperature Recovery Factor:

The variations of indicated free air temperature with airspeed were obtained from the pacer airspeed calibration data. Test data indicated that 100 percent adiabatic temperature rise was obtained with the thermocouple type bulb located on the under side of the fuselage 95 inches to the rear of the nose air intake. The expression for determining the percentage of adiabatic temperature rise is developed in Reference No. 6 and is as follows:

$$K = \left(\frac{T_{ic}}{T_{at}} - 1 \right) 5/M^2$$

11. Landings:

Normal and maximum performance landings were conducted to determine the minimum distance from a 50 foot obstacle to stop under different condtions of

operation. The data obtained have been corrected to standard day, sea level, no wind conditions by the following expressions which are presented in Reference No. 2:

$$S_{ts} = S_{as} + S_{gs}$$

$$S_{as} = S_{at} + V_{wt}$$

$$S_{gs} = S_{gt} \left(\frac{V_{td} + V_w}{V_{td}} \right)^{1.85} x\ \sigma$$

12. Stability and Control:

a. Stability and control data were obtained to evaluate the handling characteristics of the aircraft at some of the conditions where the compliance with USAF Specification was estimated to be marginal or unsatisfactory. These data are presented in Figures 28 through 94 of Appendix I. Certain limitations of instrumentations and simplifying assumption for data analysis were tolerated in the analysis of these data and are worth noting:

b. Instrumentation was not dynamically balanced.

c. Aileron roll rates were obtained from a Schwein rate gyro and the peak values obtained are presented. Since only the left aileron was instrumented, the right aileron was assumed to have the same deflection but in an opposite direction.

d. Stick force gradients have been calculated from the slopes of the stick force versus normal acceleration curves. These gradients have been used to compare test results with the requirements presented in Specification 1815-B. It has been observed that this procedure was not in strict accordance with the method outlined in 1815-B, which defines the gradient as the ratio of the force to the "g" at the test point. The 1815-B criteria were used for comparison since they generally represented the desired condition.

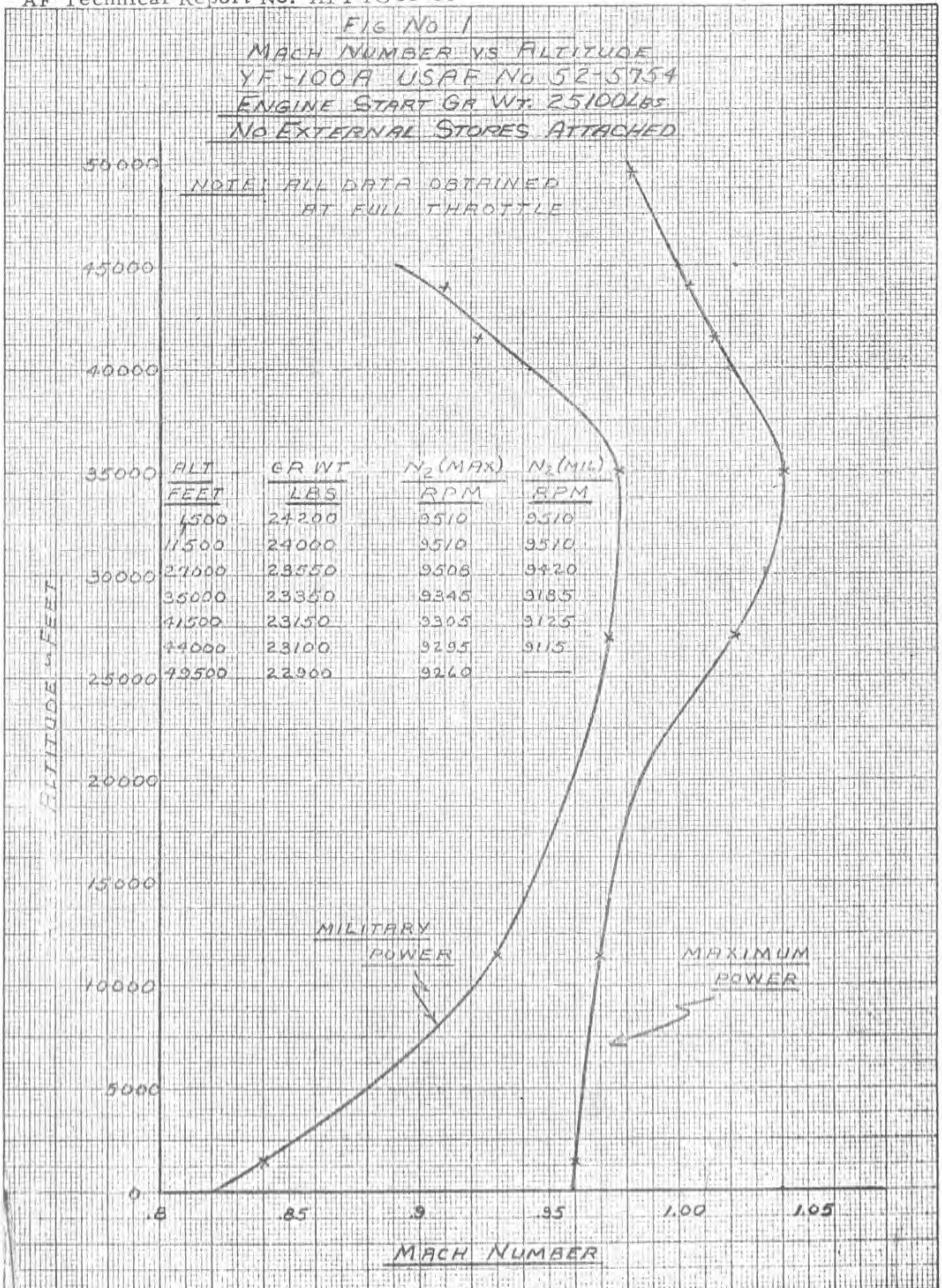


FIG NO 1
MACH NUMBER VS ALTITUDE
YF-100A USAF NO 52-5754
ENGINE START GR WT. 25100LBS
NO EXTERNAL STORES ATTACHED
NOTE: ALL DATA OBTAINED AT FULL THROTTLE
ALT FEET | GR WT LBS | N_2 (MAX) RPM | N_2 (MIL) RPM
1500 | 24200 | 9510 | 9510
11500 | 24000 | 9510 | 9510
27000 | 23550 | 9506 | 9420
35000 | 23350 | 9345 | 9185
41500 | 23150 | 9305 | 9125
44000 | 23100 | 9295 | 9115
49500 | 22900 | 9260 | —
MILITARY POWER
MAXIMUM POWER
50000
45000
40000
35000
30000
25000
20000
15000
10000
5000
0
ALTITUDE - FEET
.8
.85
.9
.95
1.00
1.05
MACH NUMBER

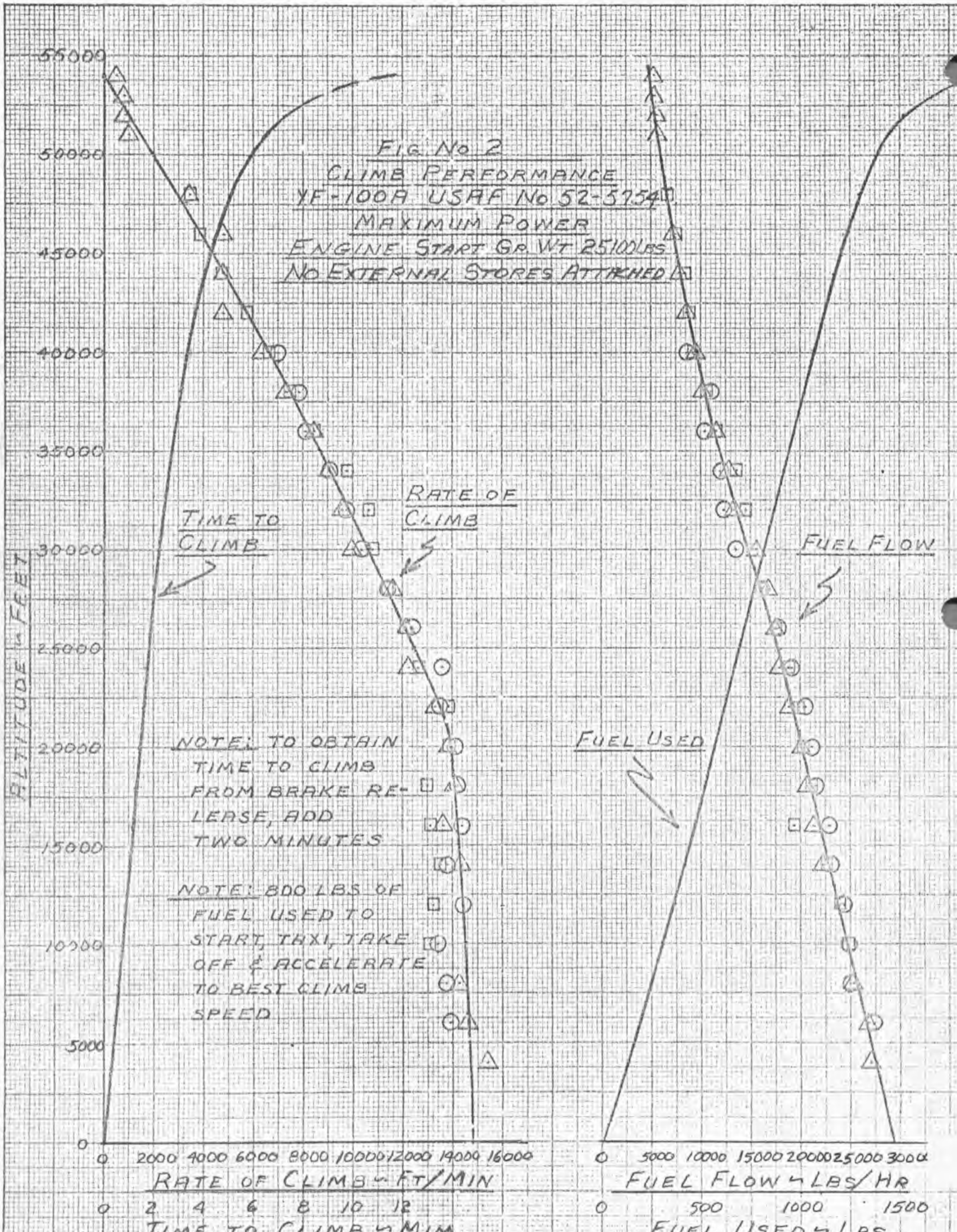
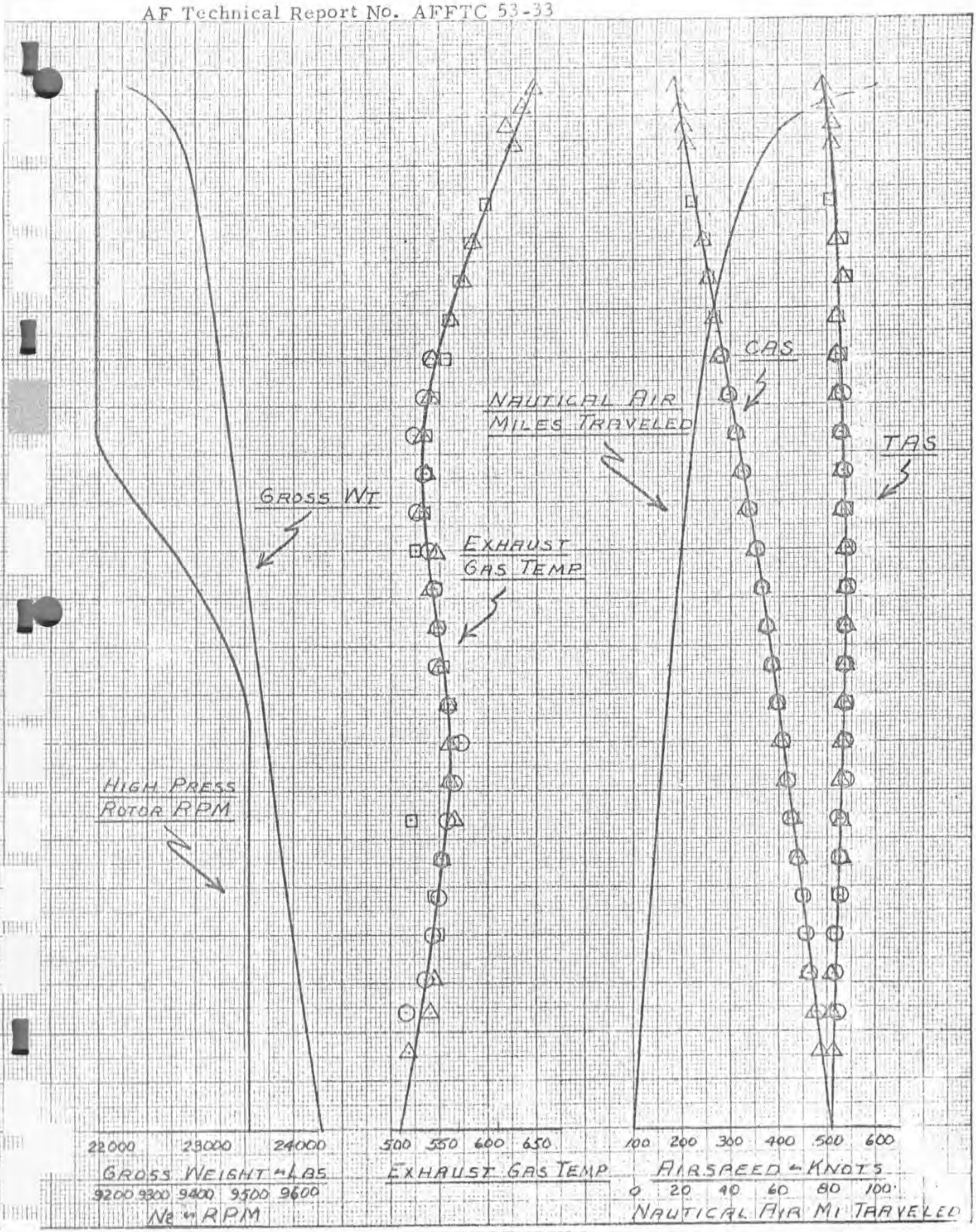


FIG NO 2
CLIMB PERFORMANCE
YF-100A USAF NO 52-5754
MAXIMUM POWER
ENGINE START GR WT 25100LBS
NO EXTERNAL STORES ATTACHED
TIME TO CLIMB
RATE OF CLIMB
FUEL FLOW
FUEL USED
NOTE: TO OBTAIN TIME TO CLIMB FROM BRAKE RELEASE, ADD TWO MINUTES
NOTE: 800 LBS OF FUEL USED TO START, TAXI, TAKE OFF & ACCELERATE TO BEST CLIMB SPEED
ALTITUDE ~ FEET
55000
50000
45000
40000
35000
30000
25000
20000
15000
10000
5000
0
0 2000 4000 6000 8000 10000 12000 14000 16000
RATE OF CLIMB ~ FT/MIN
0 2 4 6 8 10 12
TIME TO CLIMB ~ MIN
0 5000 10000 15000 20000 25000 30000
FUEL FLOW ~ LBS/HR
0 500 1000 1500
FUEL USED ~ LBS



GROSS WT
HIGH PRESS ROTOR RPM
EXHAUST GAS TEMP
NAUTICAL AIR MILES TRAVELED
CAS
TAS
22000 23000 24000
GROSS WEIGHT ~ LBS
9200 9300 9400 9500 9600
N_2 ~ RPM
500 550 600 650
EXHAUST GAS TEMP
100 200 300 400 500 600
AIRSPEED ~ KNOTS
0 20 40 60 80 100
NAUTICAL AIR MI TRAVELED

APPENDIX I

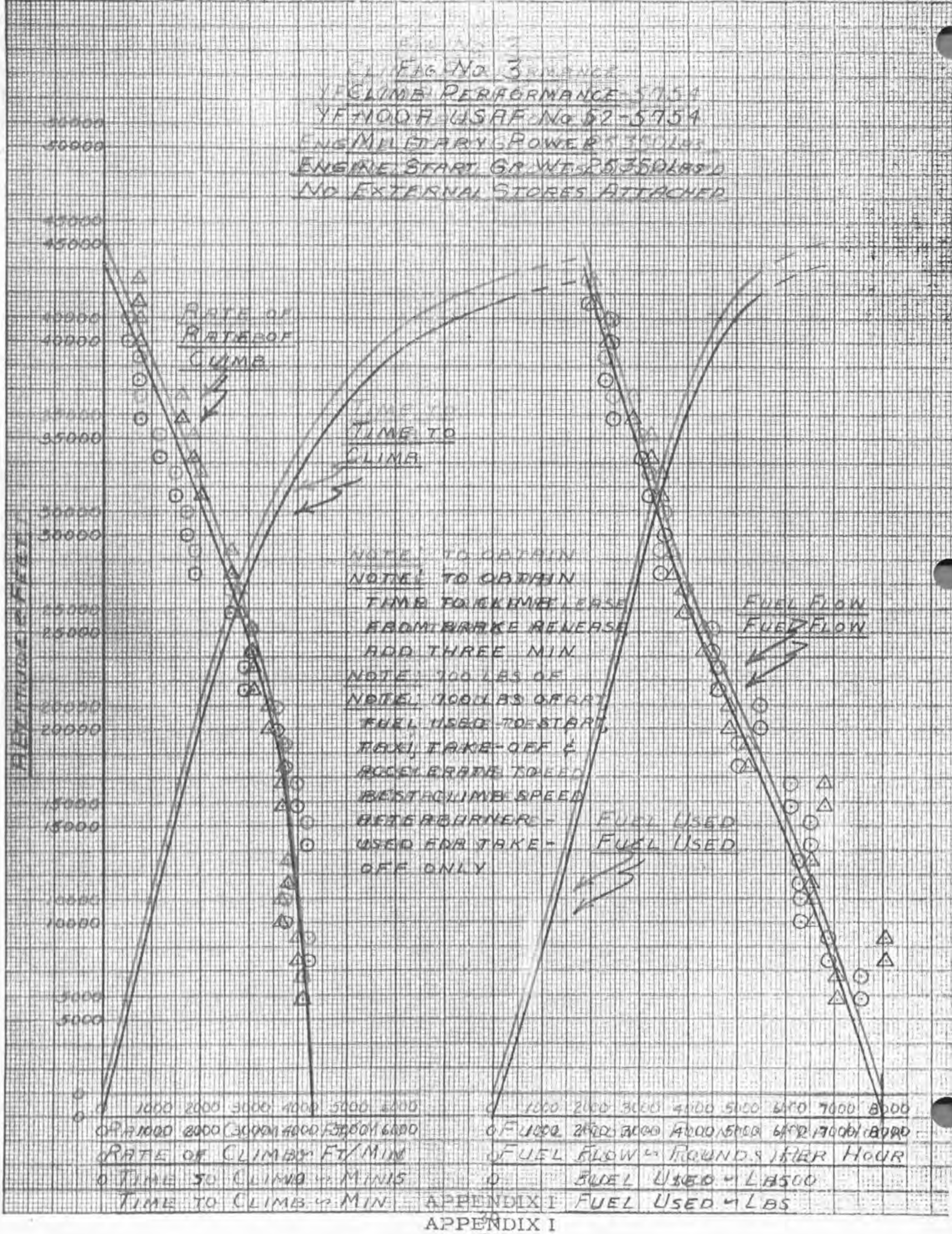
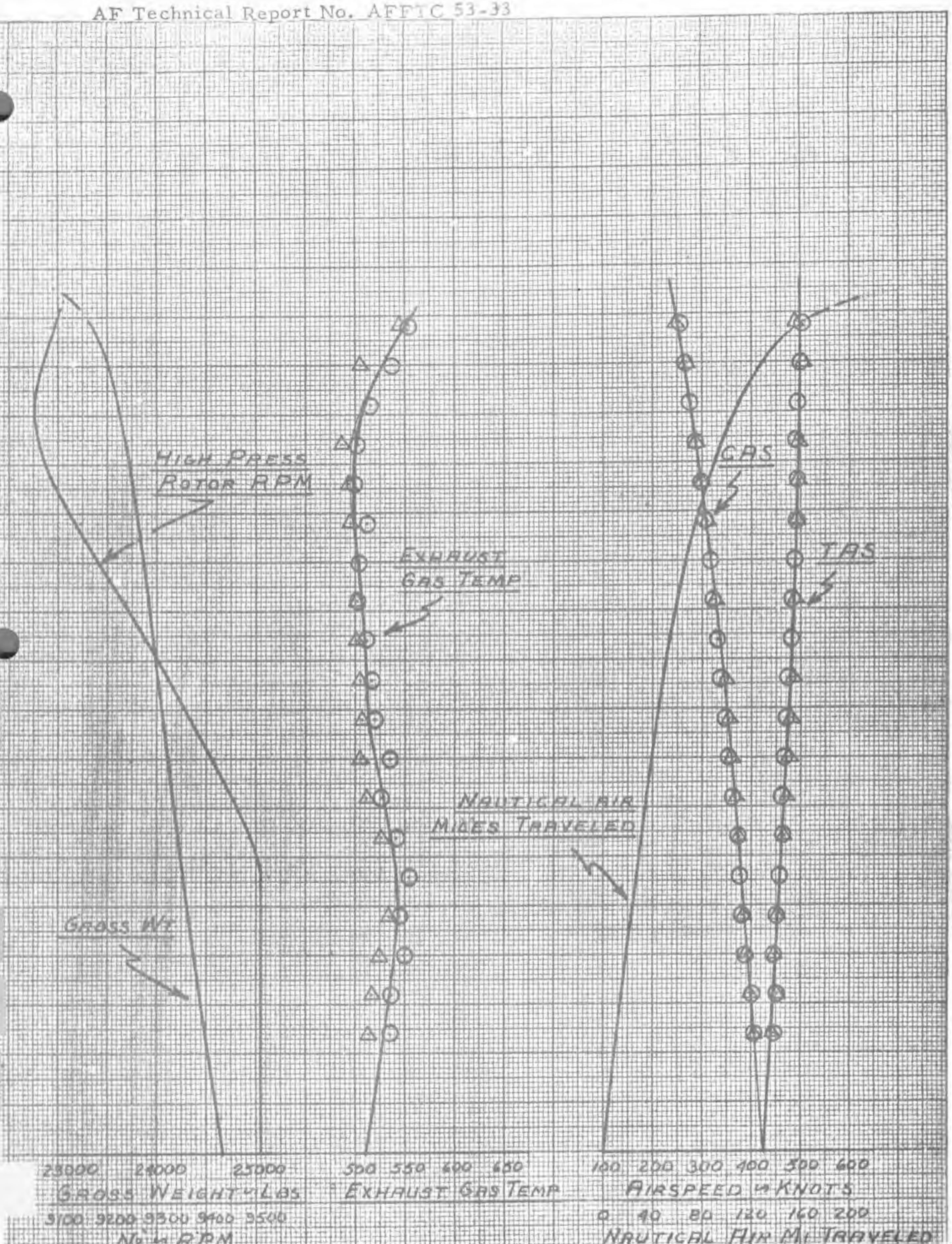


FIG No 3
CLIMB PERFORMANCE
YF-100A USAF No 52-5754
MILITARY POWER
ENGINE START GR WT 25350 LBS
NO EXTERNAL STORES ATTACHED
RATE OF CLIMB
TIME TO CLIMB
NOTE: TO OBTAIN TIME TO CLIMB FROM BRAKE RELEASE ADD THREE MIN
NOTE: 700 LBS OF FUEL USED TO START, TAXI, TAKE-OFF & ACCELERATE TO BEST CLIMB SPEED. AFTERBURNER USED FOR TAKE-OFF ONLY
FUEL FLOW
FUEL USED
ALTITUDE ~ FEET
45000
40000
35000
30000
25000
20000
15000
10000
5000
0
0 1000 2000 3000 4000 5000 6000
RATE OF CLIMB ~ FT/MIN
TIME TO CLIMB ~ MIN
0 1000 2000 3000 4000 5000 6000 7000 8000
FUEL FLOW ~ POUNDS PER HOUR
FUEL USED ~ LBS


HIGH PRESS ROTOR RPM
GROSS WT
EXHAUST GAS TEMP
NAUTICAL AIR MILES TRAVELED
CAS
TAS
23000 24000 25000
GROSS WEIGHT ~ LBS
9100 9200 9300 9400 9500
N2 ~ RPM
500 550 600 650
EXHAUST GAS TEMP
100 200 300 400 500 600
AIRSPEED ~ KNOTS
0 40 80 120 160 200
NAUTICAL AIR MI TRAVELED

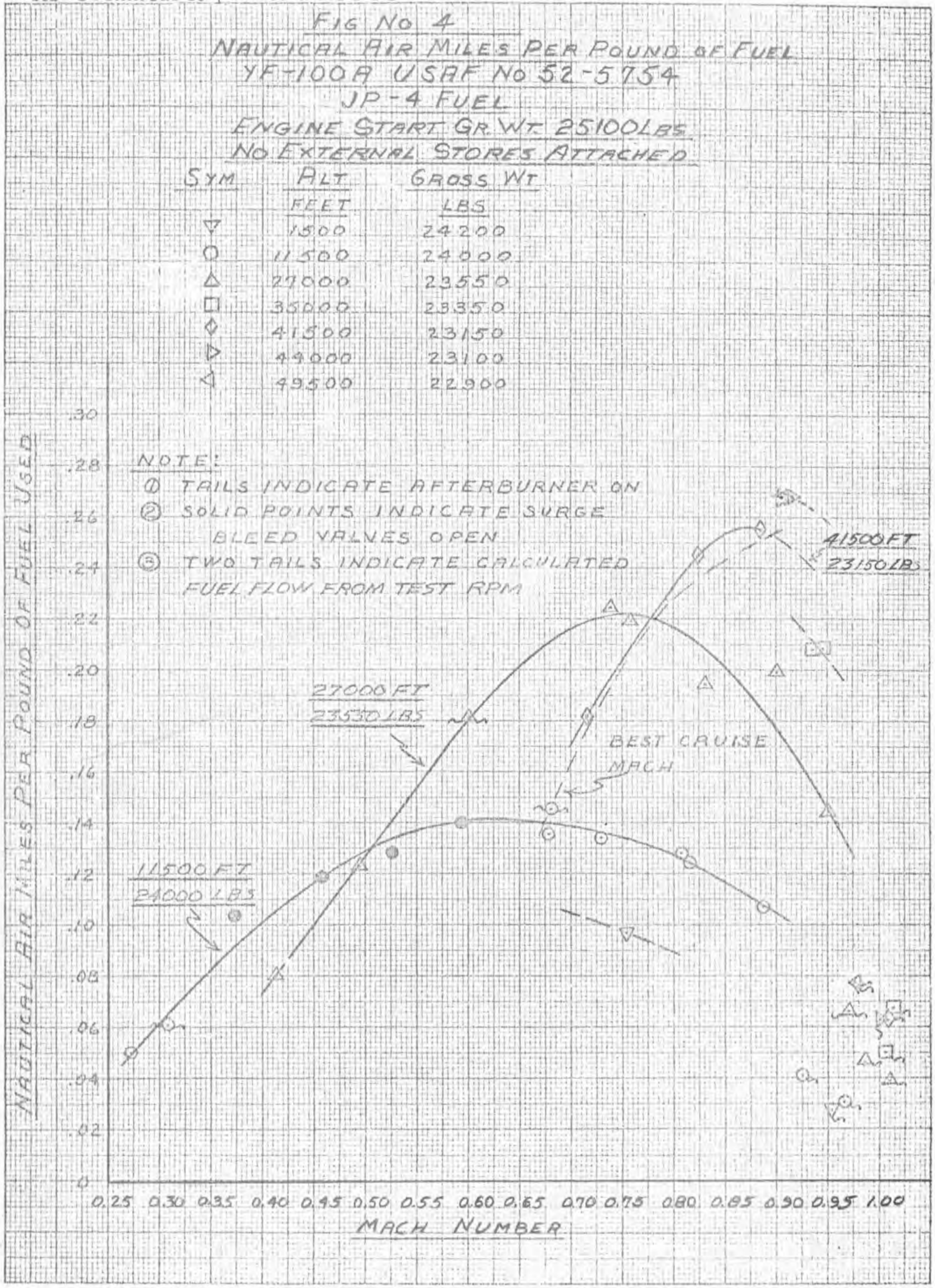


FIG NO 4
NAUTICAL AIR MILES PER POUND OF FUEL
YF-100A USAF NO 52-5754
JP-4 FUEL
ENGINE START GR WT 25100LBS
NO EXTERNAL STORES ATTACHED
SYM ALT GROSS WT
FEET LBS
1500 24200
11500 24000
27000 23550
35000 23350
41500 23150
49000 23100
49500 22900
NOTE:
① TAILS INDICATE AFTERBURNER ON
② SOLID POINTS INDICATE SURGE BLEED VALVES OPEN
③ TWO TAILS INDICATE CALCULATED FUEL FLOW FROM TEST RPM
41500 FT
23150 LBS
27000 FT
23550 LBS
BEST CRUISE MACH
11500 FT
24000 LBS
NAUTICAL AIR MILES PER POUND OF FUEL USED
.30
.28
.26
.24
.22
.20
.18
.16
.14
.12
.10
.08
.06
.04
.02
0
0.25 0.30 0.35 0.40 0.45 0.50 0.55 0.60 0.65 0.70 0.75 0.80 0.85 0.90 0.95 1.00
MACH NUMBER

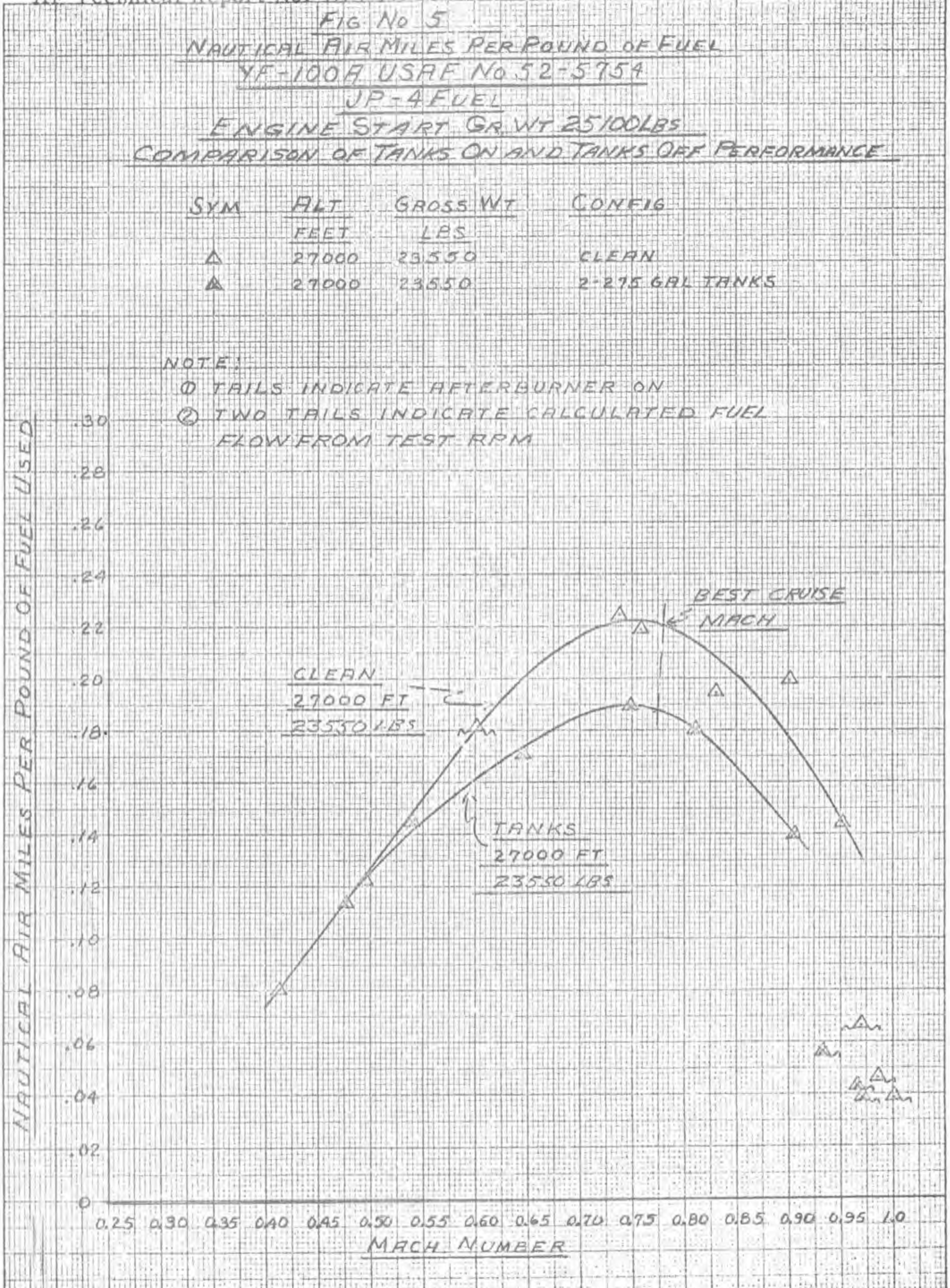


FIG No 5
NAUTICAL AIR MILES PER POUND OF FUEL
YF-100A USAF No 52-5754
JP-4 FUEL
ENGINE START GR WT 25100 LBS
COMPARISON OF TANKS ON AND TANKS OFF PERFORMANCE
SYM ALT FEET GROSS WT LBS CONFIG
27000 23550 CLEAN
27000 23550 2-275 GAL TANKS
NOTE:
① TAILS INDICATE AFTERBURNER ON
② TWO TAILS INDICATE CALCULATED FUEL FLOW FROM TEST RPM
NAUTICAL AIR MILES PER POUND OF FUEL USED
.30
.28
.26
.24
.22
.20
.18
.16
.14
.12
.10
.08
.06
.04
.02
0
BEST CRUISE MACH
CLEAN
27000 FT
23550 LBS
TANKS
27000 FT
23550 LBS
0.25 0.30 0.35 0.40 0.45 0.50 0.55 0.60 0.65 0.70 0.75 0.80 0.85 0.90 0.95 1.0
MACH NUMBER

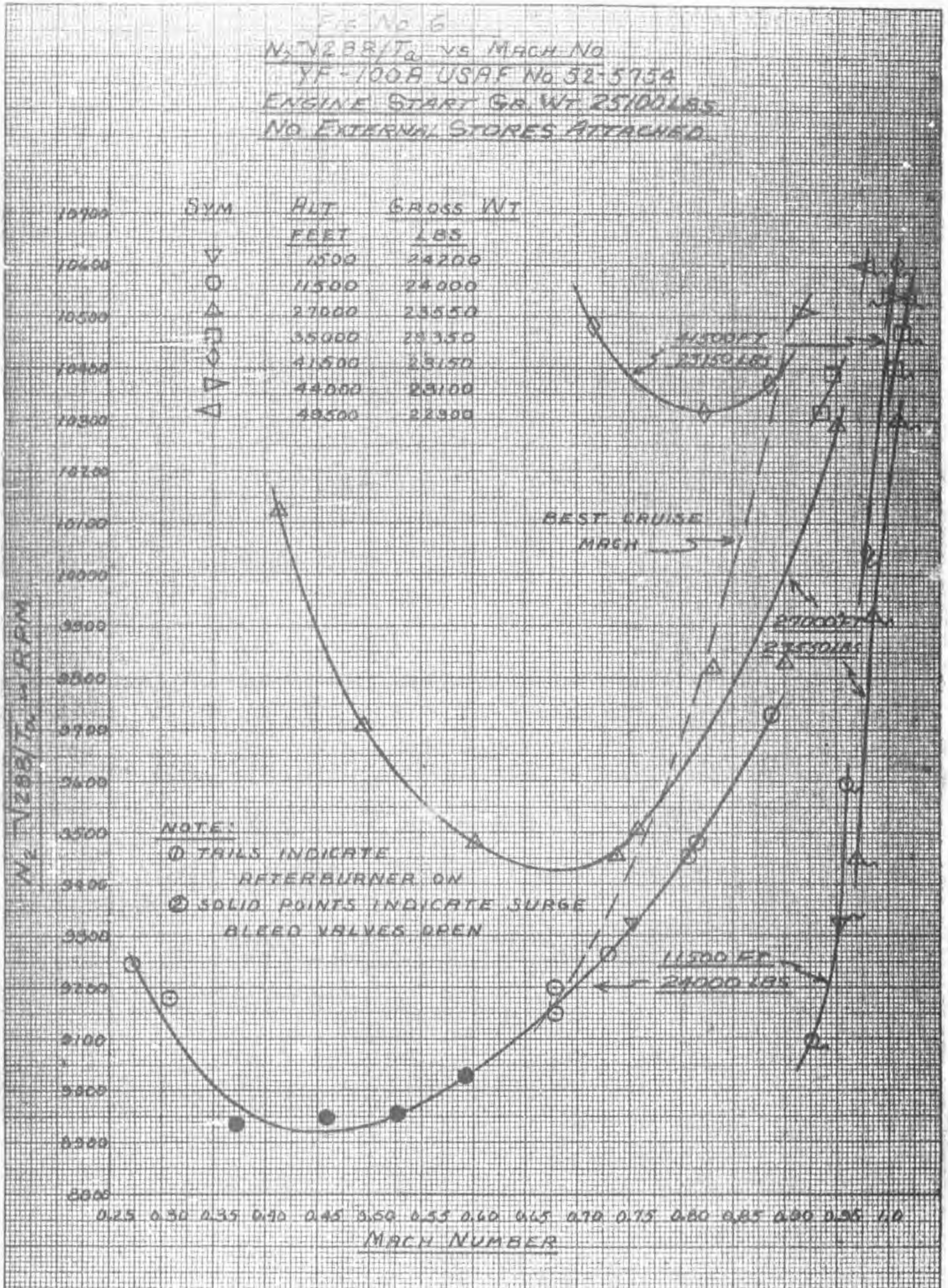


FIG No 6
$N_2\sqrt{288/T_a}$ VS MACH NO
YF-100A USAF No 52-5754
ENGINE START GR. WT 25100 LBS
NO EXTERNAL STORES ATTACHED
SYM
ALT FEET
GROSS WT LBS
1500 24200
11500 24000
27000 23550
35000 23350
41500 23150
44000 23100
48500 22900
41500 FT 23150 LBS
BEST CRUISE MACH
27000 FT 23550 LBS
11500 FT 24000 LBS
NOTE:
① TAILS INDICATE AFTERBURNER ON
② SOLID POINTS INDICATE SURGE BLEED VALVES OPEN
$N_2\sqrt{288/T_a}$ - RPM
10700
10600
10500
10400
10300
10200
10100
10000
9900
9800
9700
9600
9500
9400
9300
9200
9100
9000
8900
8800
0.25 0.30 0.35 0.40 0.45 0.50 0.55 0.60 0.65 0.70 0.75 0.80 0.85 0.90 0.95 1.0
MACH NUMBER

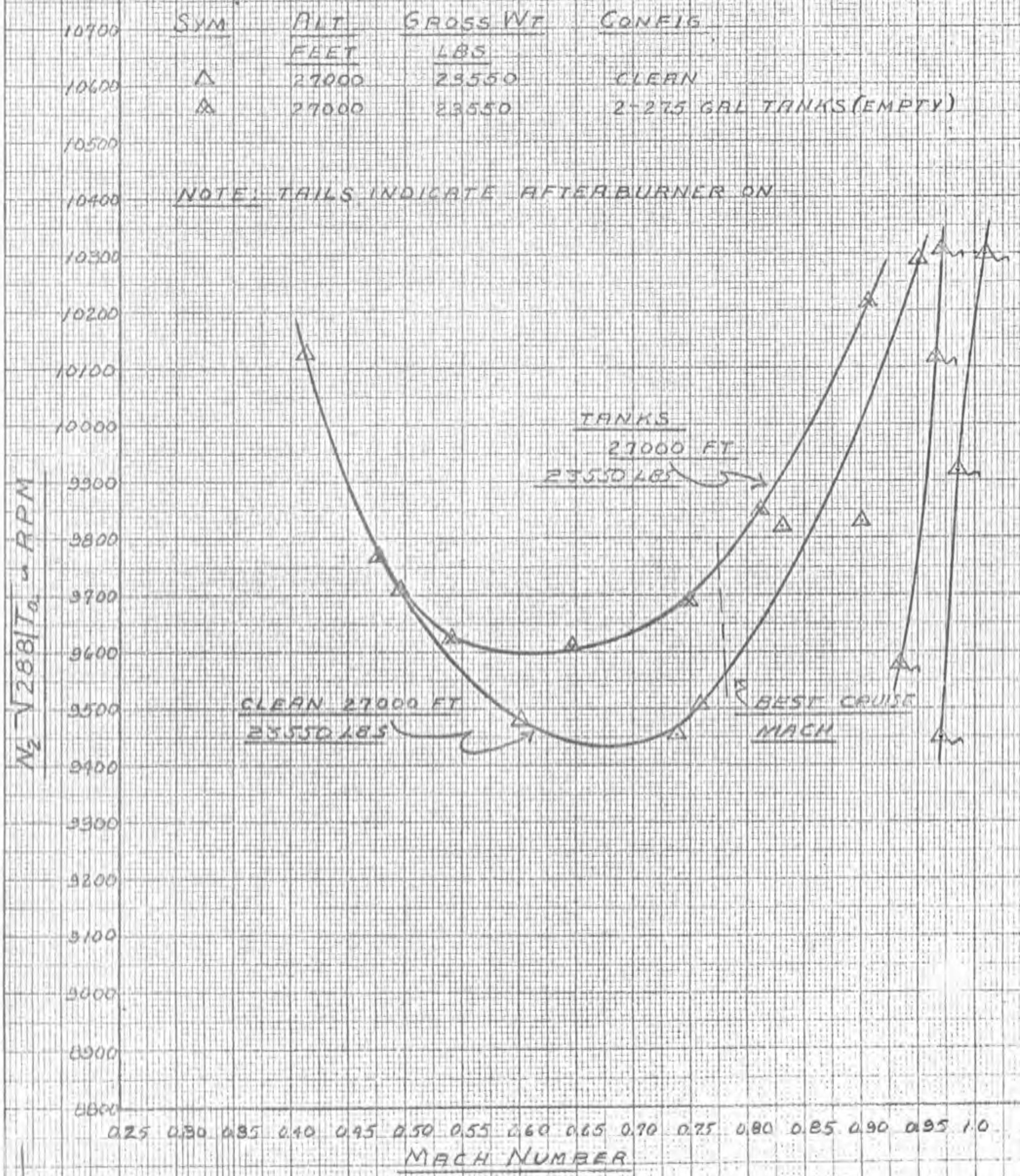
FIG NO 7

$N_2\sqrt{288/T_a}$ VS MACH NO.

YF-100A USAF NO 52-5754

ENGINE START GR WT 25100 LBS

COMPARISON OF TANKS ON AND TANKS OFF PERFORMANCE



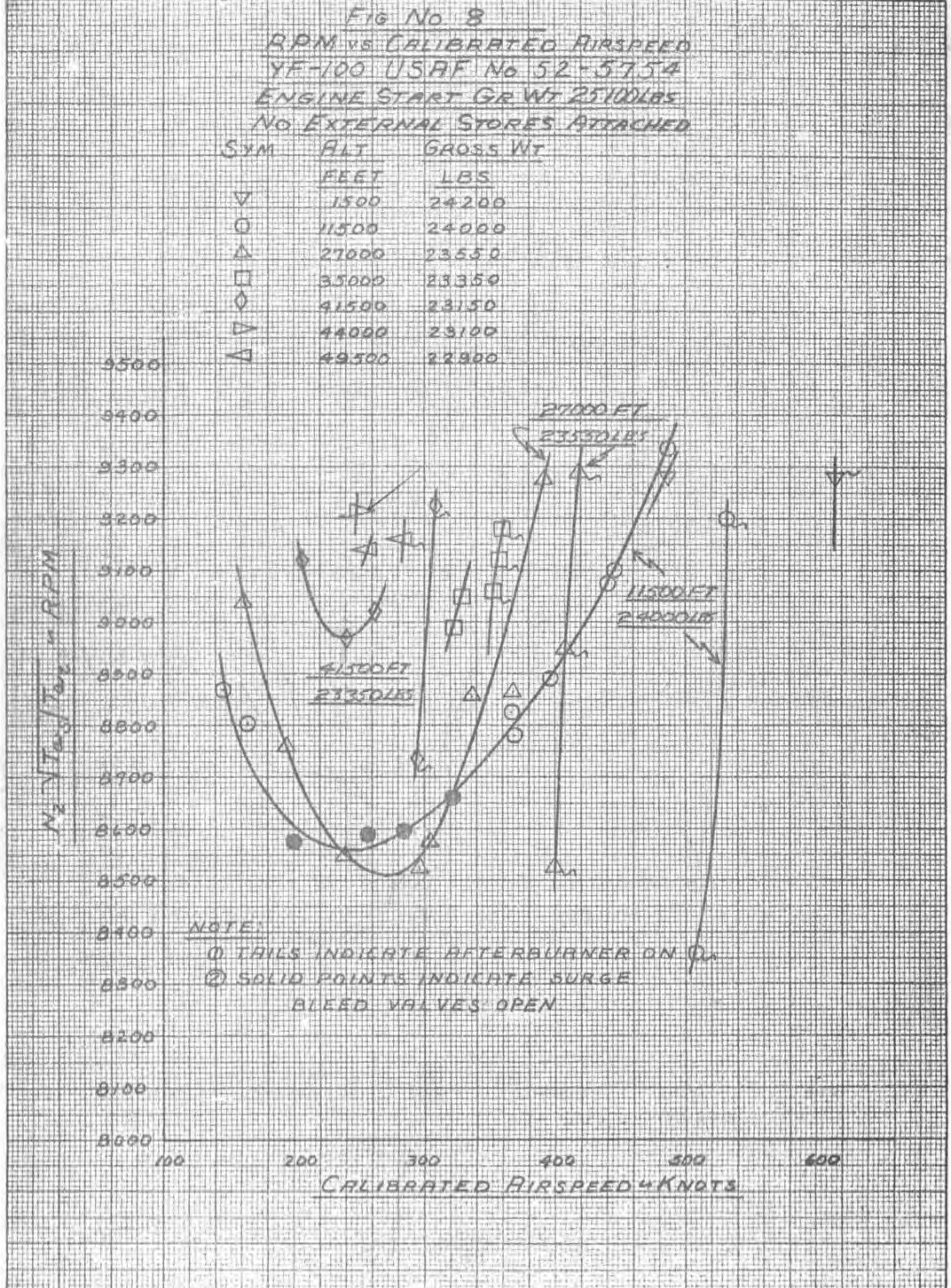


FIG NO 8
RPM VS CALIBRATED AIRSPEED
YF-100 USAF NO 52-5754
ENGINE START GR WT 25100 LBS
NO EXTERNAL STORES ATTACHED
SYM ALT GROSS WT
FEET LBS
1500 24200
11500 24000
27000 23550
35000 23350
41500 23150
44000 23100
49500 22900
27000 FT
23550 LBS
11500 FT
24000 LBS
41500 FT
23350 LBS
NOTE:
① TAILS INDICATE AFTERBURNER ON
② SOLID POINTS INDICATE SURGE BLEED VALVES OPEN
CALIBRATED AIRSPEED ~ KNOTS
9500
9400
9300
9200
9100
9000
8900
8800
8700
8600
8500
8400
8300
8200
8100
8000
100
200
300
400
500
600

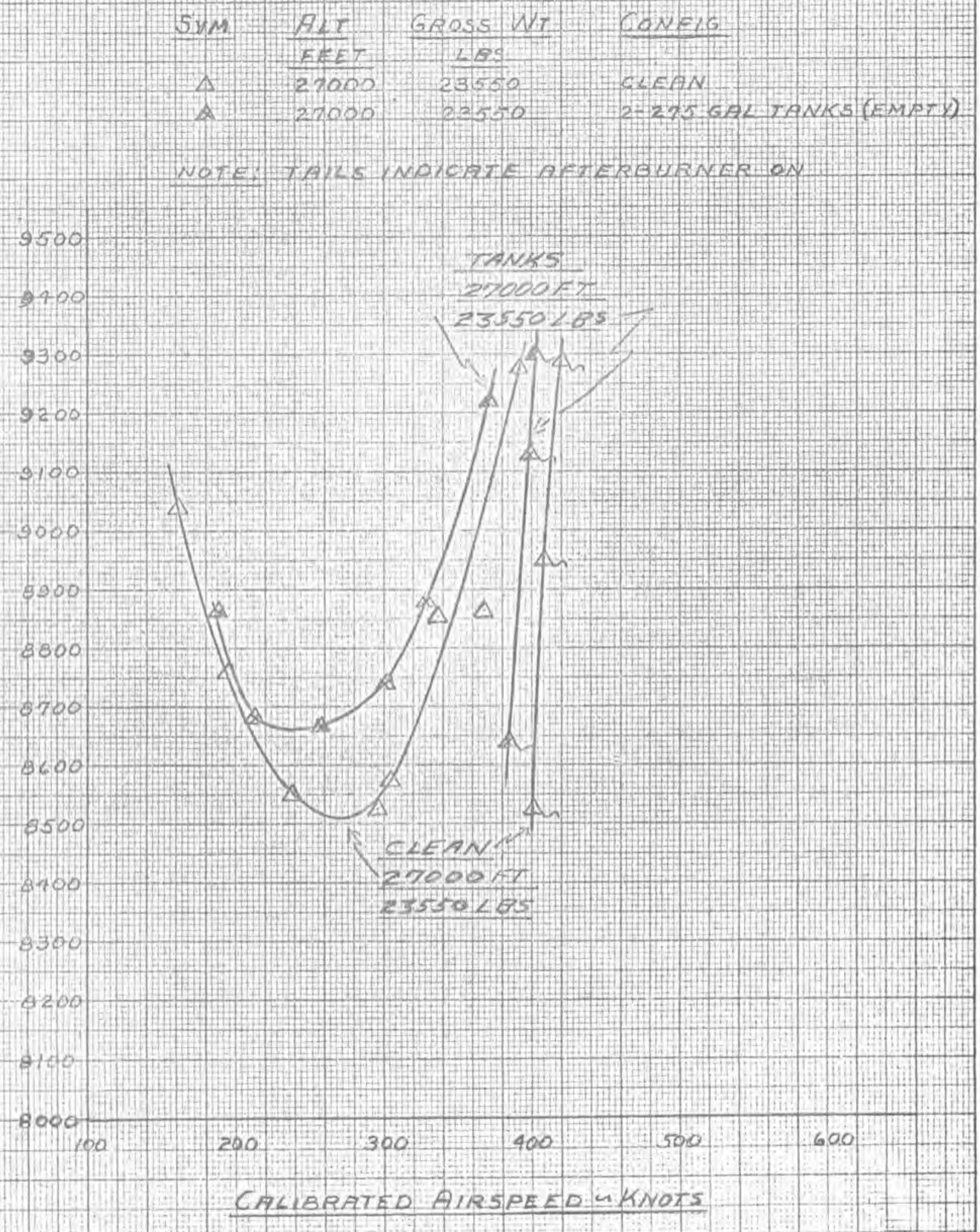
FIG NO 9

RPM VS CALIBRATED AIRSPEED

YF-100A USAF No 52-5754

ENGINE START GROSS WT 25100 LBS

COMPARISON OF TANKS ON AND TANKS OFF PERFORMANCE



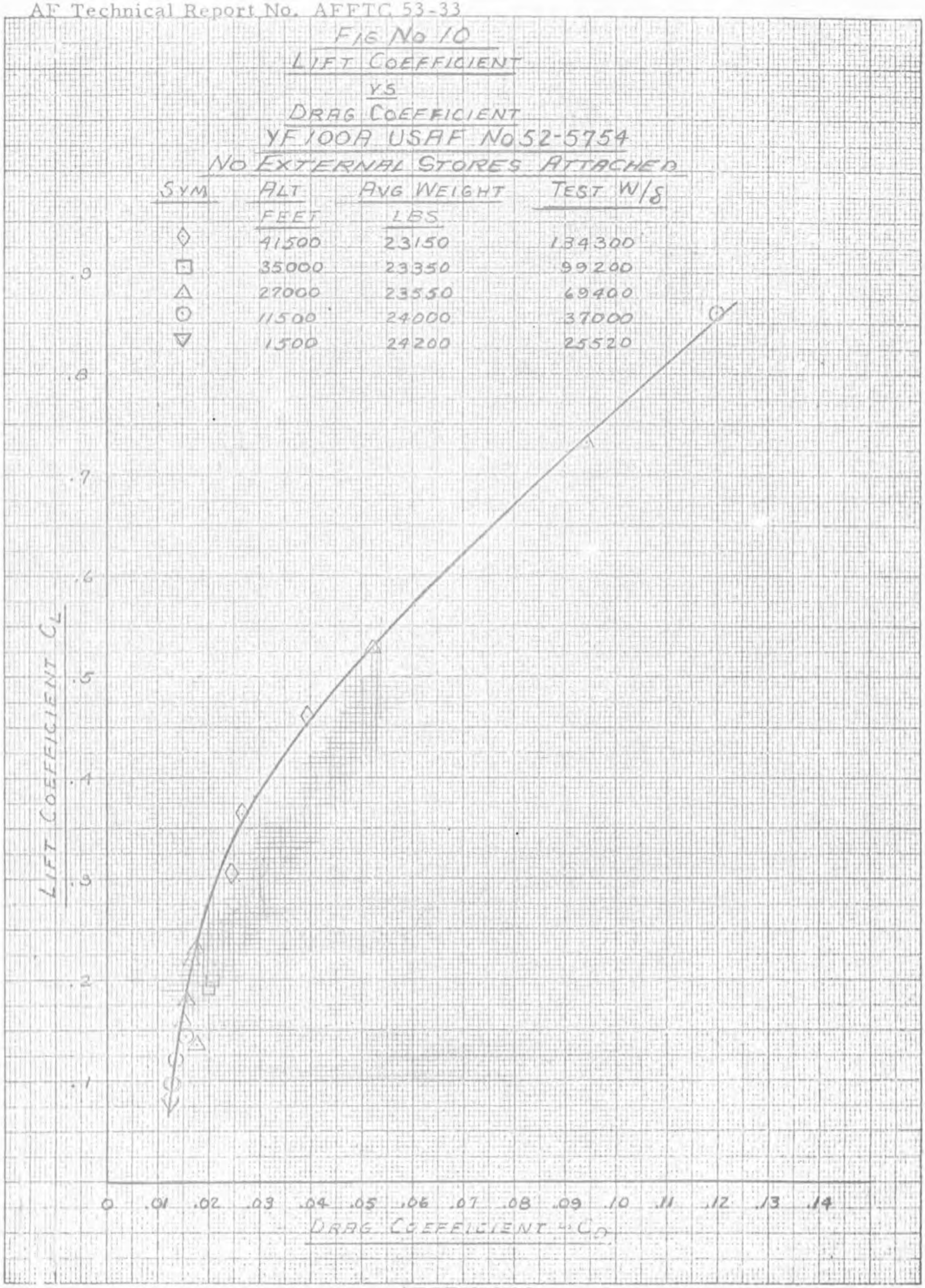


FIG No 10
LIFT COEFFICIENT
VS
DRAG COEFFICIENT
YF100A USAF No 52-5754
NO EXTERNAL STORES ATTACHED
SYM
ALT FEET
AVG WEIGHT LBS
TEST W/δ
41500
23150
134300
35000
23350
99200
27000
23550
69400
11500
24000
37000
1500
24200
25520
LIFT COEFFICIENT C_L
.9
.8
.7
.6
.5
.4
.3
.2
.1
0
.01
.02
.03
.04
.05
.06
.07
.08
.09
.10
.11
.12
.13
.14
DRAG COEFFICIENT – C_D

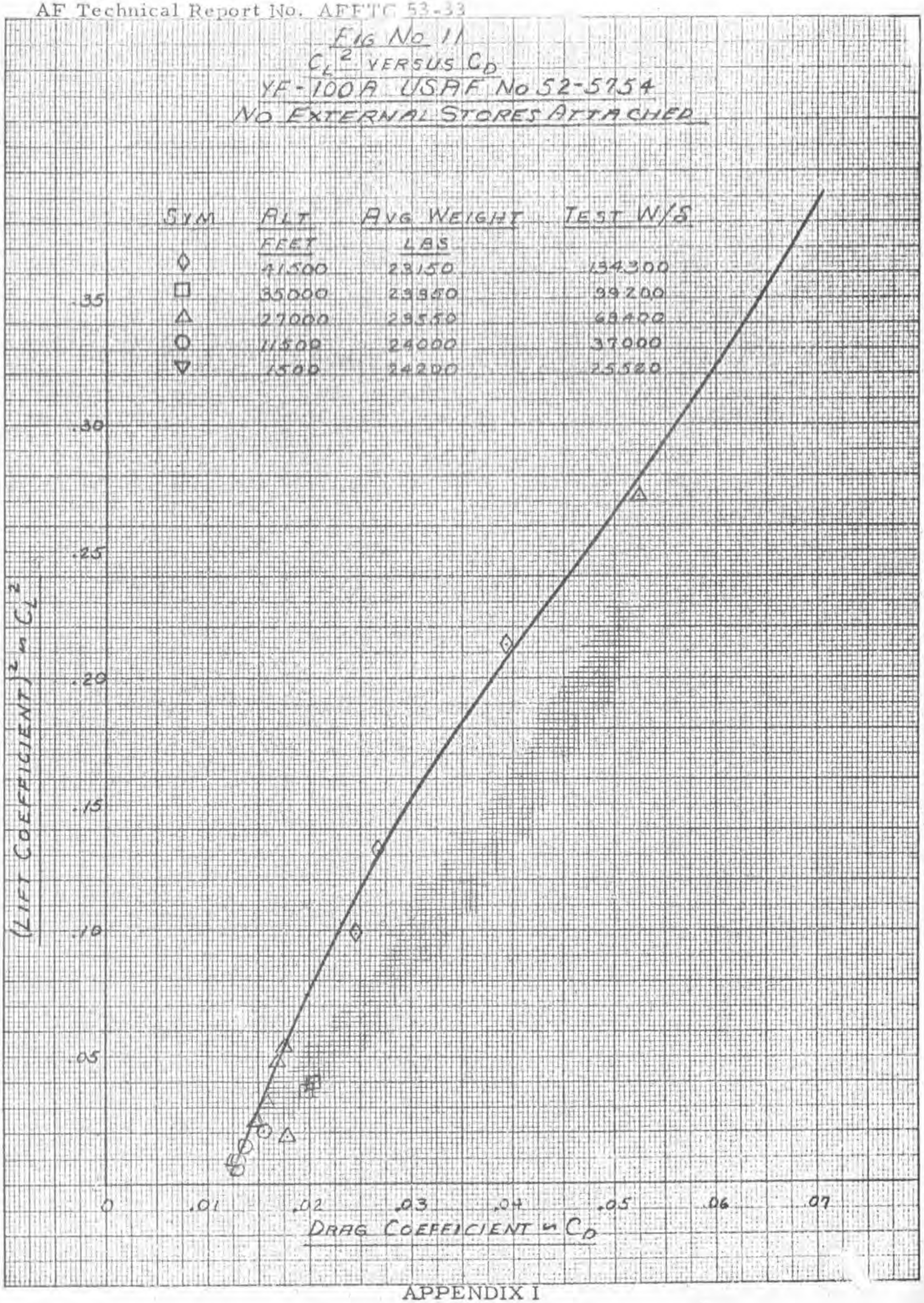


FIG NO 11
C_L^2 VERSUS C_D
YF-100A USAF No 52-5754
NO EXTERNAL STORES ATTACHED
SYM
ALT FEET
AVG WEIGHT LBS
TEST W/S
41500
23150
134300
35000
23950
99200
27000
23550
68400
11500
24000
37000
1500
24200
25580
.35
.30
.25
.20
.15
.10
.05
0
.01
.02
.03
.04
.05
.06
.07
(LIFT COEFFICIENT)^2 ~ C_L^2
DRAG COEFFICIENT ~ C_D

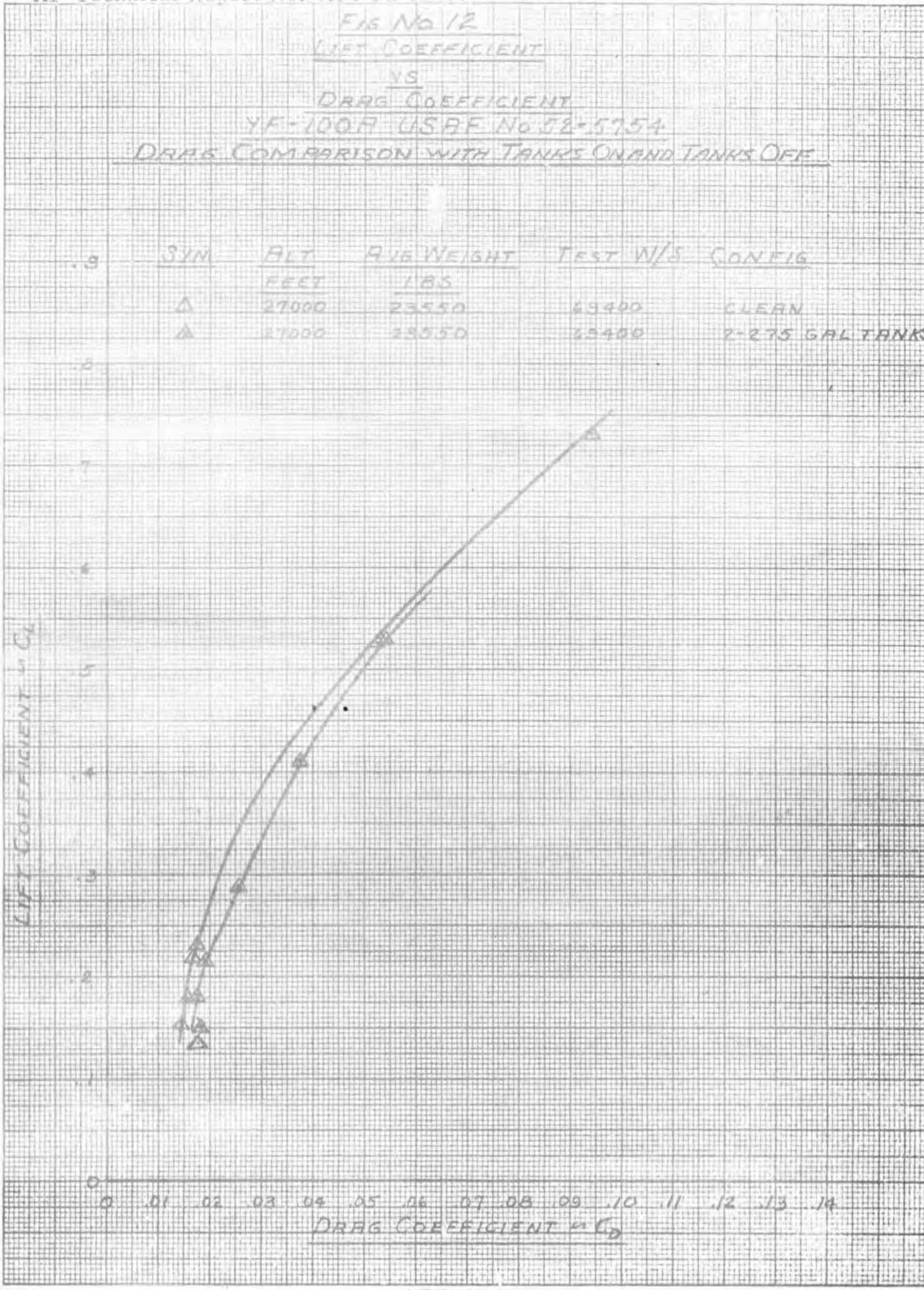


FIG No 12
LIFT COEFFICIENT
VS
DRAG COEFFICIENT
YF-100A USAF No 52-5754
DRAG COMPARISON WITH TANKS ON AND TANKS OFF
SYM ALT FEET AVG WEIGHT LBS TEST W/S CONFIG
27000 23550 65400 CLEAN
27000 23550 65400 2-275 GAL TANKS
LIFT COEFFICIENT ~ C_L
.9 .8 .7 .6 .5 .4 .3 .2 .1 0
0 .01 .02 .03 .04 .05 .06 .07 .08 .09 .10 .11 .12 .13 .14
DRAG COEFFICIENT ~ C_D

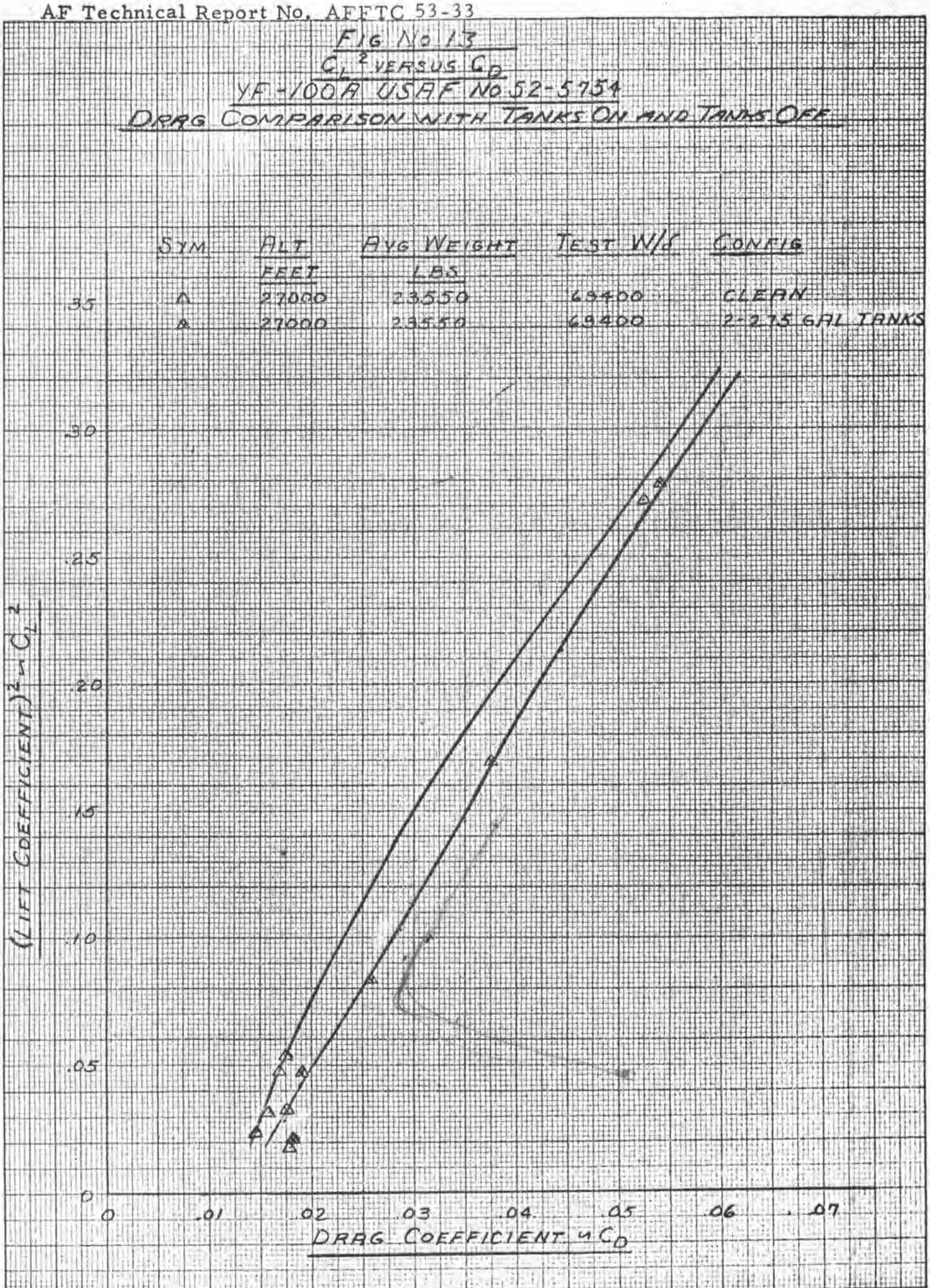


FIG NO 13
C_L^2 VERSUS C_D
YF-100A USAF NO 52-5754
DRAG COMPARISON WITH TANKS ON AND TANKS OFF
SYM | ALT FEET | AVG WEIGHT LBS | TEST W/δ | CONFIG
△ | 27000 | 23550 | 69400 | CLEAN
▲ | 27000 | 23550 | 69400 | 2-275 GAL TANKS
.35
.30
.25
.20
.15
.10
.05
0
(LIFT COEFFICIENT)^2 ~ C_L^2
0
.01
.02
.03
.04
.05
.06
.07
DRAG COEFFICIENT ~ C_D

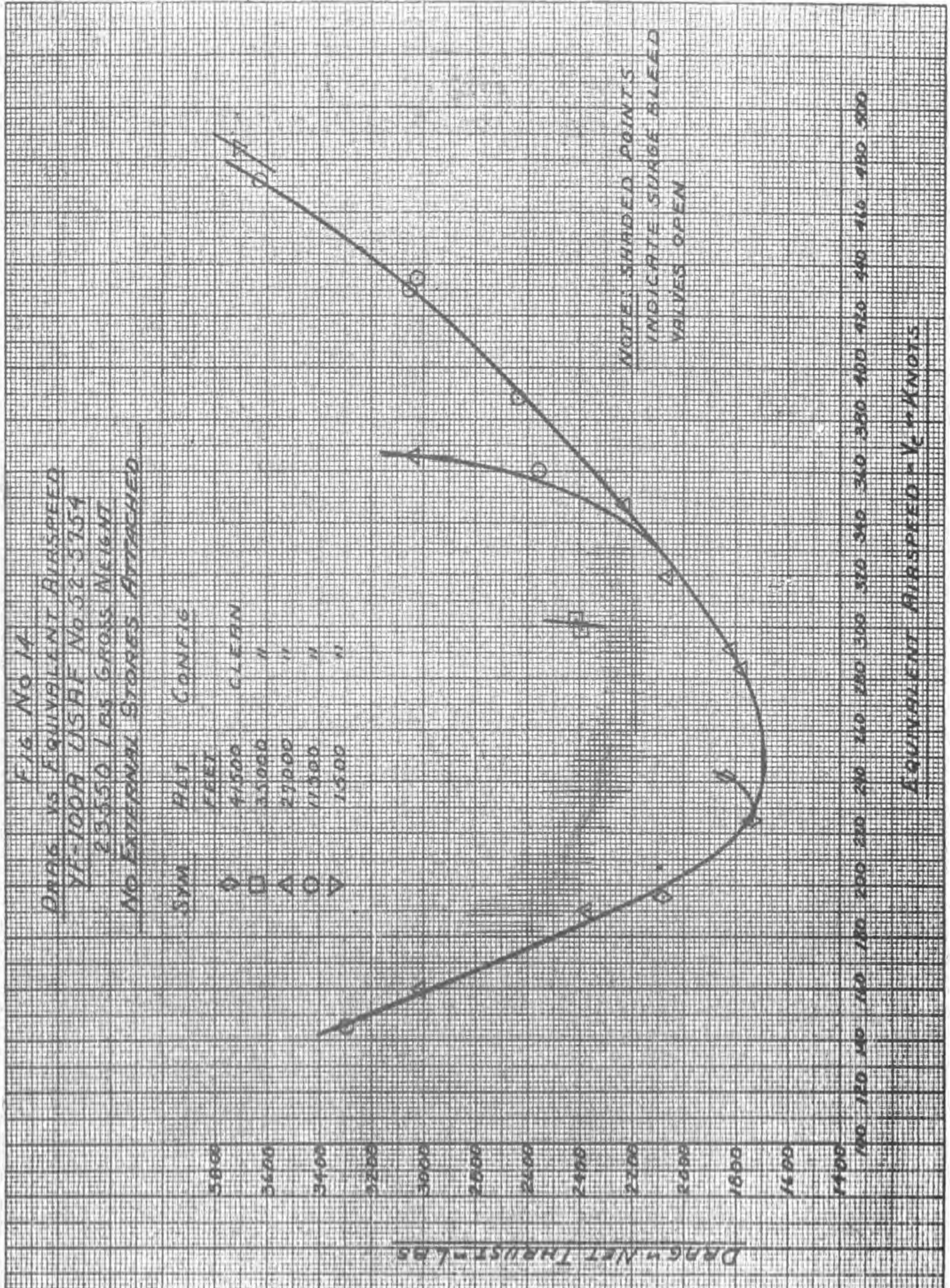


FIG No 14
DRAG VS EQUIVALENT AIRSPEED
YF-100A USAF No 52-5754
23550 LBS GROSS WEIGHT
NO EXTERNAL STORES ATTACHED
SYM ALT FEET CONFIG
41500 CLEAN
35000 "
27000 "
11300 "
1500 "
NOTE: SHADED POINTS INDICATE SURGE BLEED VALVES OPEN
DRAG = NET THRUST - LBS
EQUIVALENT AIRSPEED - Ve - KNOTS

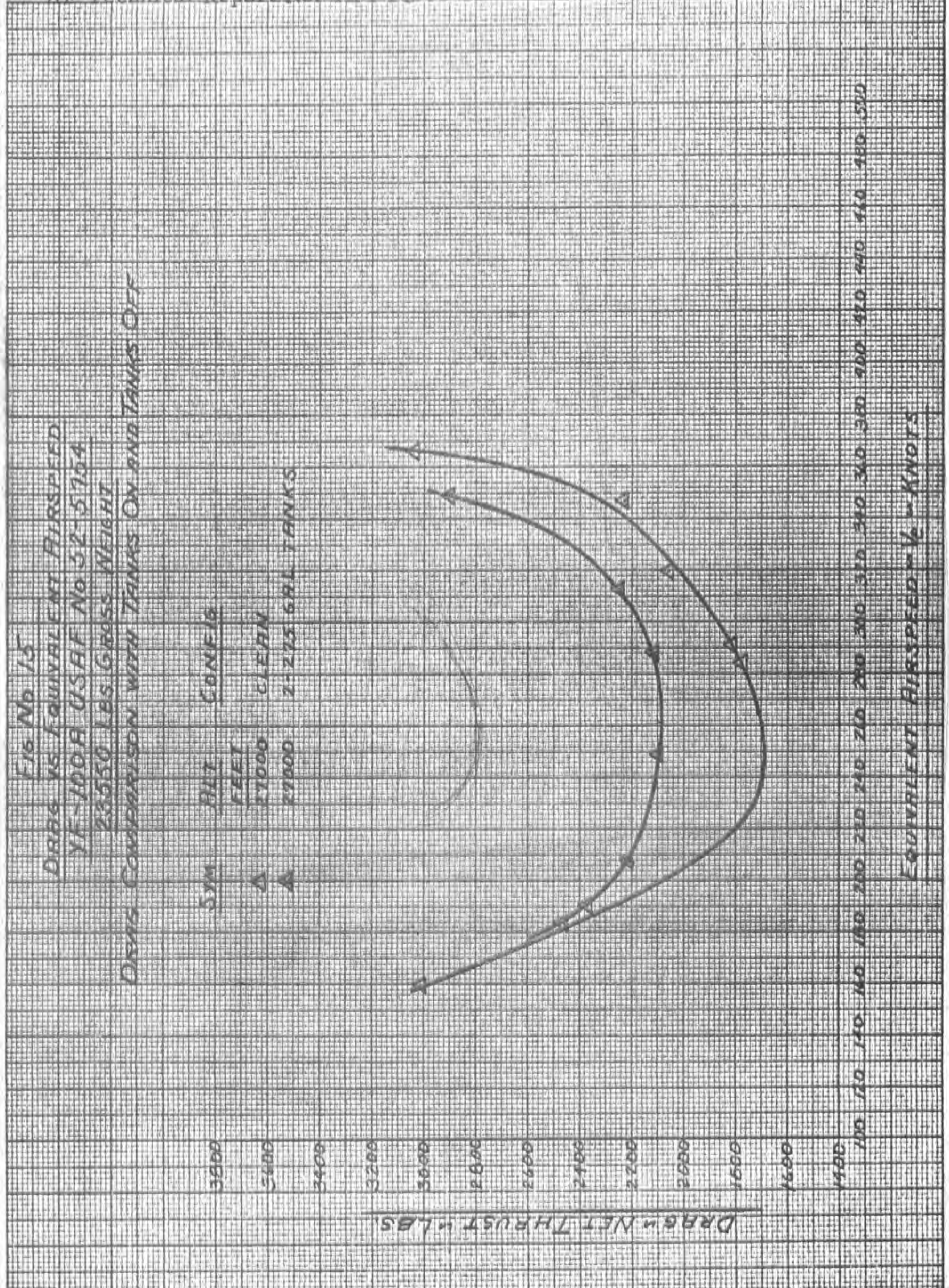


FIG NO 15
DRAG VS EQUIVALENT AIRSPEED
YF-100A USAF No 52-5754
23550 LBS GROSS WEIGHT
DRAG COMPARISON WITH TANKS ON AND TANKS OFF
SYM ALT FEET CONFIG
27000 CLEAN
27000 2-275 GAL TANKS
DRAG = NET THRUST ~ LBS
EQUIVALENT AIRSPEED ~ Ve ~ KNOTS

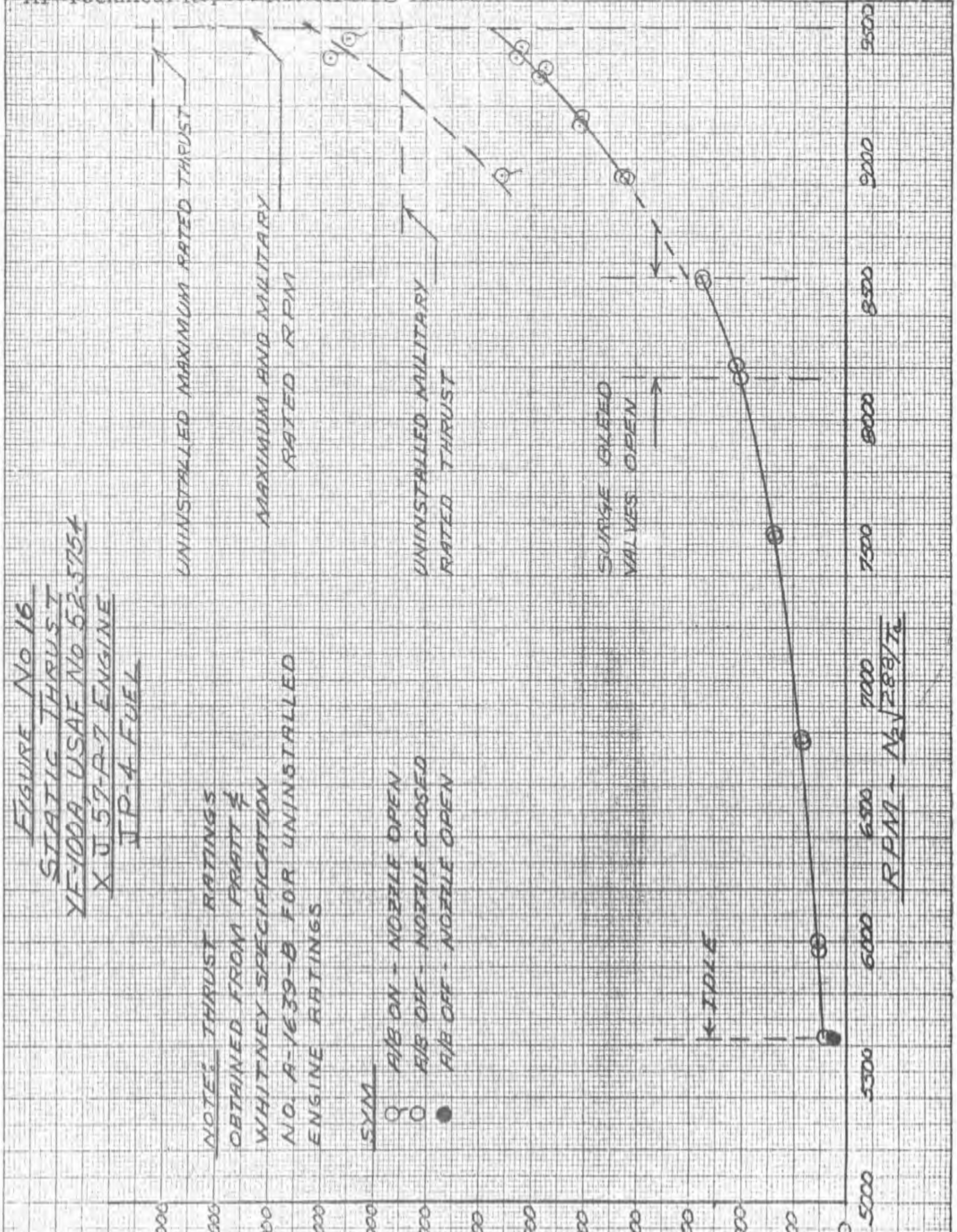


FIGURE No 16
STATIC THRUST
YF-100A, USAF No 52-5754
XJ57-P-7 ENGINE
JP-4 FUEL
NOTE: THRUST RATINGS OBTAINED FROM PRATT & WHITNEY SPECIFICATION NO. A-1639-B FOR UNINSTALLED ENGINE RATINGS
SYM
A/B ON - NOZZLE OPEN
A/B OFF - NOZZLE CLOSED
A/B OFF - NOZZLE OPEN
UNINSTALLED MAXIMUM RATED THRUST
MAXIMUM AND MILITARY RATED RPM
UNINSTALLED MILITARY RATED THRUST
SURGE BLEED VALVES OPEN
IDLE
THRUST ~ Fg 29.92/Pa ~ POUNDS
RPM ~ N2√288/T2
0
1000
2000
3000
4000
5000
6000
7000
8000
9000
10000
11000
12000
13000
5000
5500
6000
6500
7000
7500
8000
8500
9000
9500

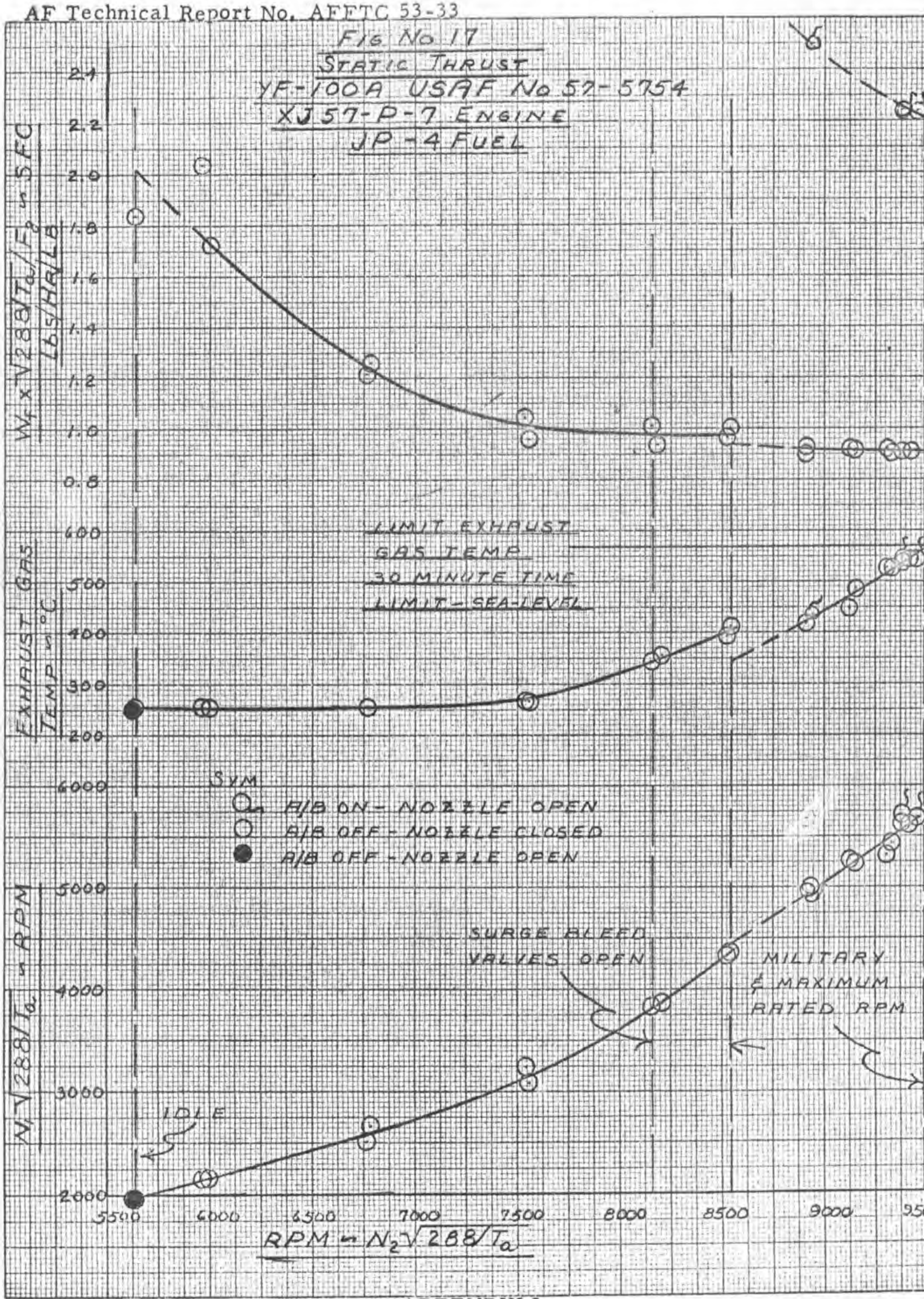


FIG NO 17
STATIC THRUST
YF-100A USAF No 52-5754
XJ57-P-7 ENGINE
JP-4 FUEL
$W_f \times \sqrt{288/T_a}/F_g$ ~ SFC
Lbs/Hr/LB
EXHAUST GAS TEMP ~ °C
$N_1\sqrt{288/T_a}$ ~ RPM
LIMIT EXHAUST GAS TEMP 30 MINUTE TIME LIMIT - SEA-LEVEL
SYM
A/B ON - NOZZLE OPEN
A/B OFF - NOZZLE CLOSED
A/B OFF - NOZZLE OPEN
SURGE BLEED VALVES OPEN
MILITARY & MAXIMUM RATED RPM
IDLE
RPM ~ $N_2\sqrt{288/T_a}$

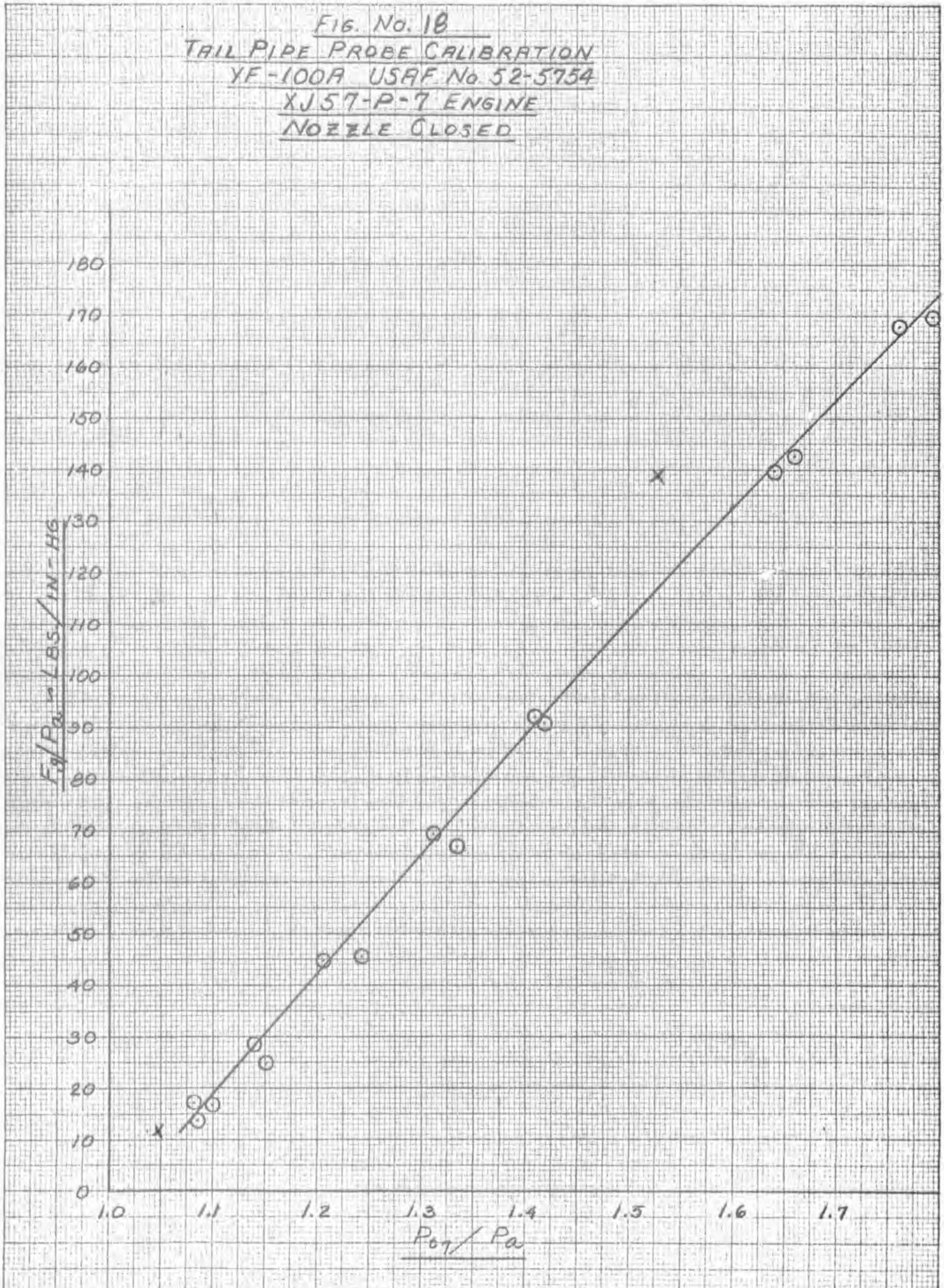


FIG. NO. 18
TAIL PIPE PROBE CALIBRATION
YF-100A USAF No. 52-5754
XJ57-P-7 ENGINE
NOZZLE CLOSED
Fg/Pa ~ LBS/IN-HG
180
170
160
150
140
130
120
110
100
90
80
70
60
50
40
30
20
10
0
1.0
1.1
1.2
1.3
1.4
1.5
1.6
1.7
Pt7/Pa

APPENDIX I

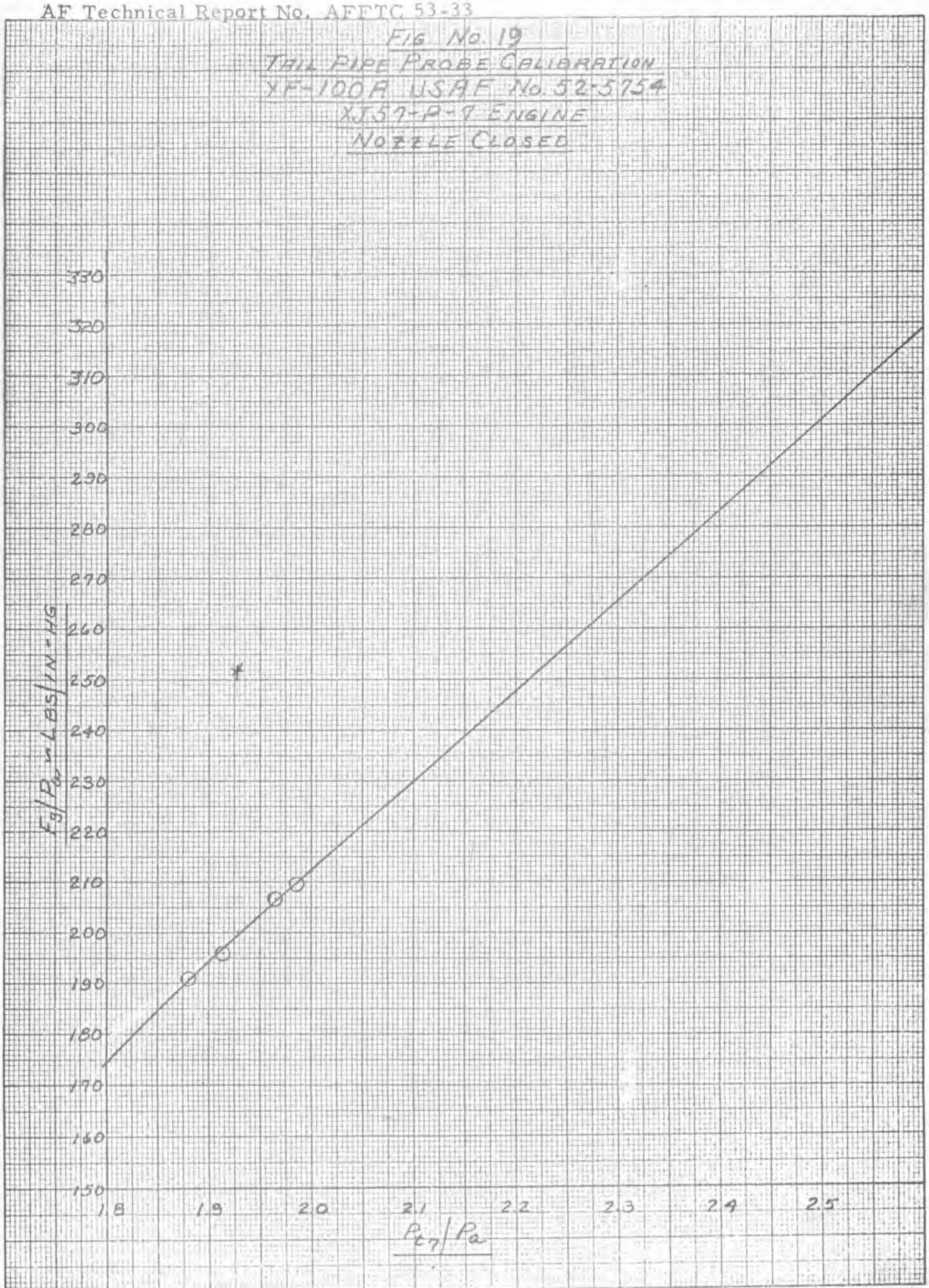


FIG. NO. 19
TAIL PIPE PROBE CALIBRATION
XF-100A USAF No. 52-5754
XJ57-P-7 ENGINE
NOZZLE CLOSED
330
320
310
300
290
280
270
260
250
240
230
220
210
200
190
180
170
160
150
F_g/P_a ~ LBS/IN-HG
1.8
1.9
2.0
2.1
2.2
2.3
2.4
2.5
P_{t_7}/P_a

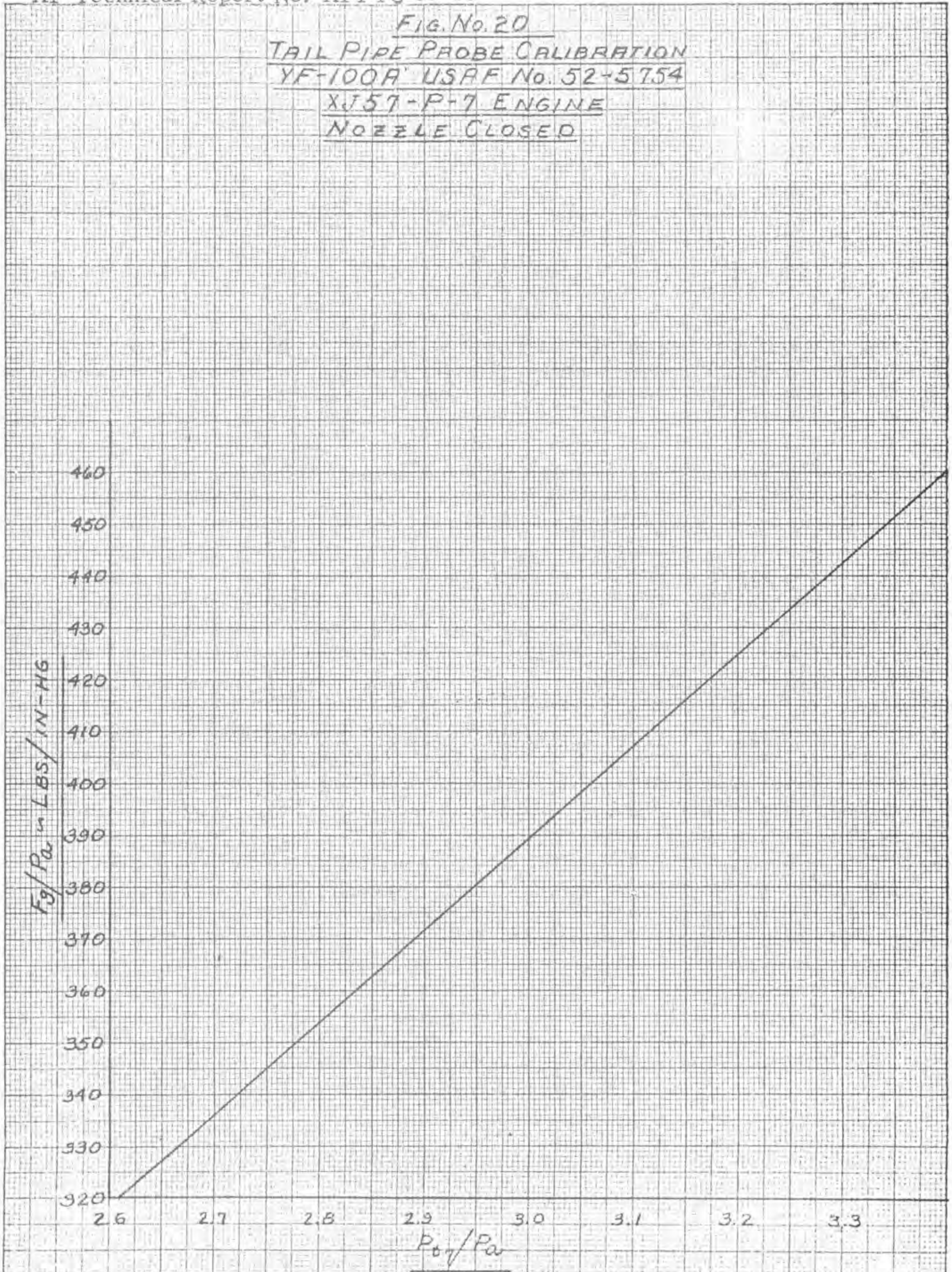


FIG. No. 20
TAIL PIPE PROBE CALIBRATION
YF-100A USAF No. 52-5754
XJ57-P-7 ENGINE
NOZZLE CLOSED
Fg/Pa ~ LBS/IN-HG
460
450
440
430
420
410
400
390
380
370
360
350
340
330
320
2.6
2.7
2.8
2.9
3.0
3.1
3.2
3.3
PtT/Pa

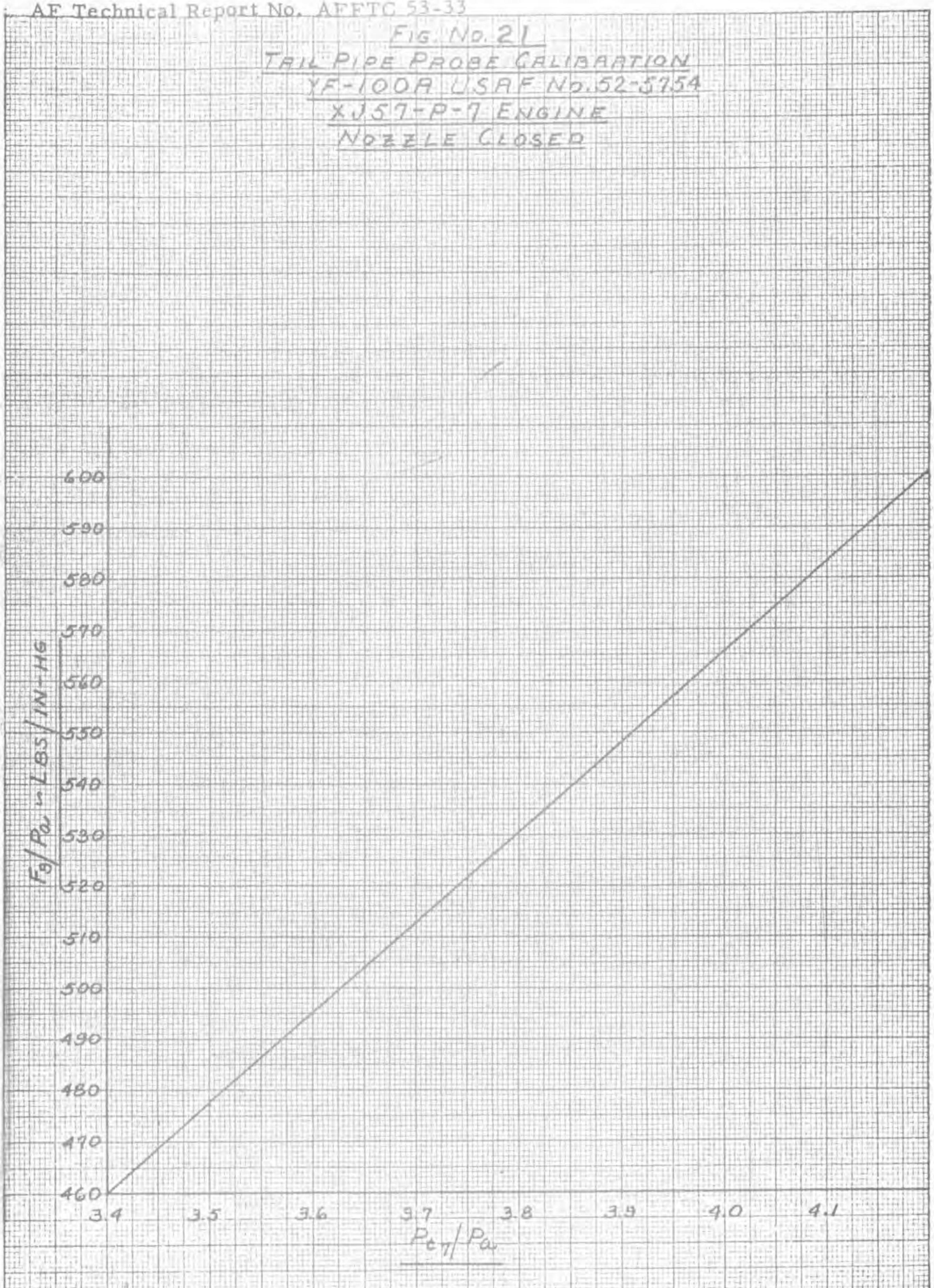


FIG. NO. 21
TAIL PIPE PROBE CALIBRATION
YF-100A USAF No. 52-5754
XJ57-P-7 ENGINE
NOZZLE CLOSED
F_g/P_a ~ LBS/IN-HG
600
590
580
570
560
550
540
530
520
510
500
490
480
470
460
3.4
3.5
3.6
3.7
3.8
3.9
4.0
4.1
P_{t_7}/P_a

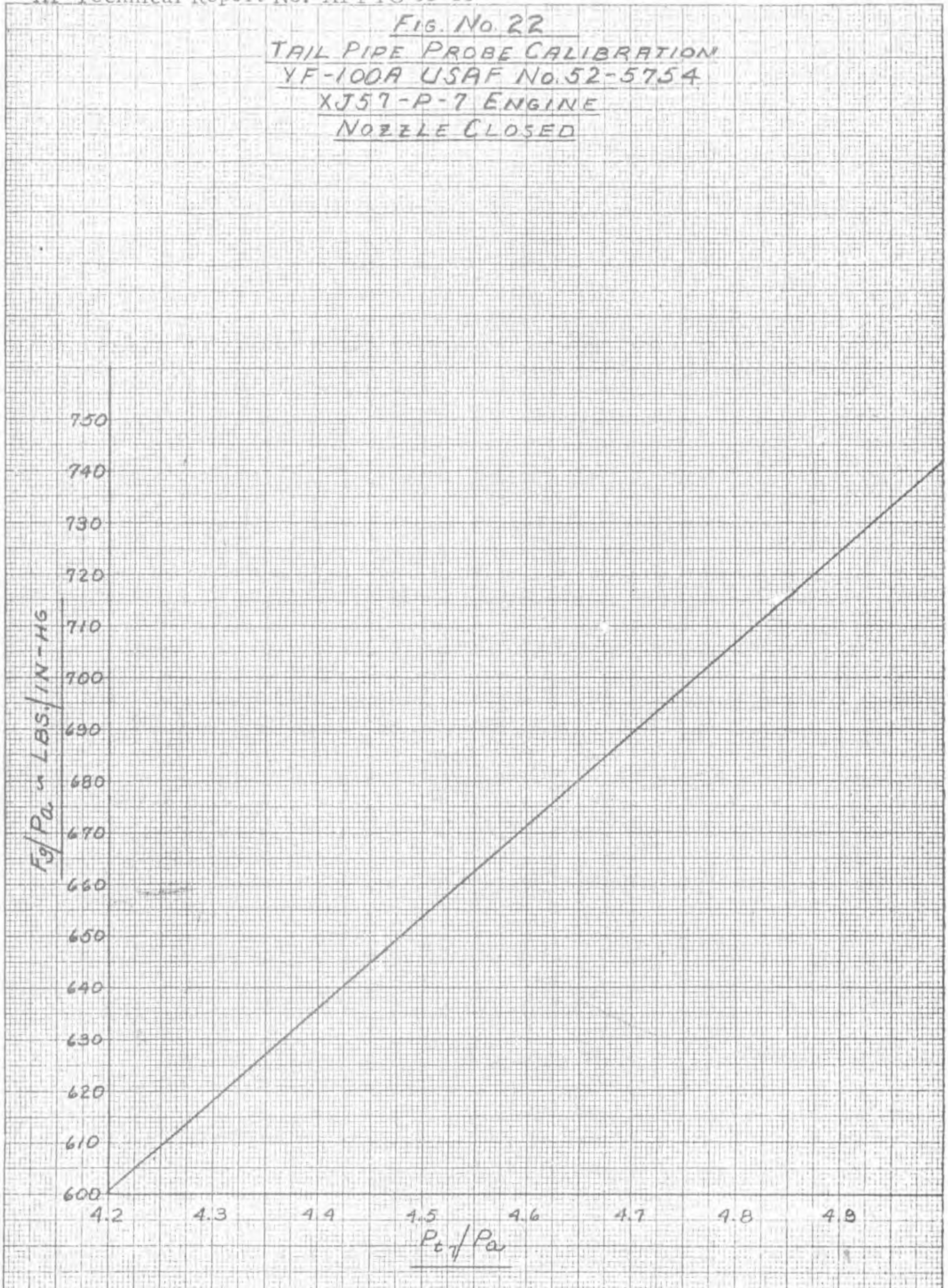


FIG. NO. 22
TAIL PIPE PROBE CALIBRATION
YF-100A USAF NO. 52-5754
XJ57-P-7 ENGINE
NOZZLE CLOSED
F_g/P_a ~ LBS./IN-HG
750
740
730
720
710
700
690
680
670
660
650
640
630
620
610
600
4.2
4.3
4.4
4.5
4.6
4.7
4.8
4.9
P_{t_7}/P_a

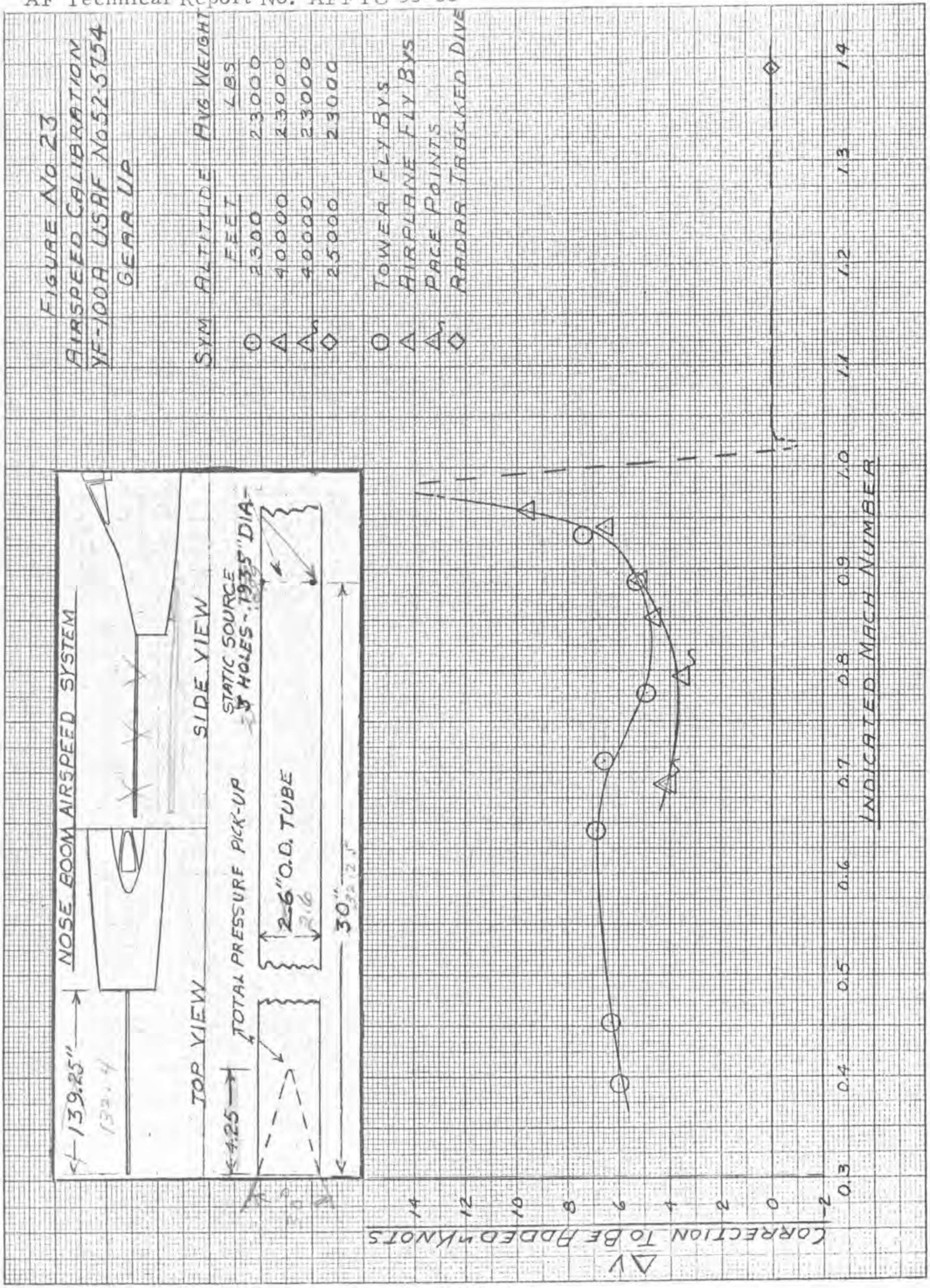


FIGURE No 23
AIRSPEED CALIBRATION
YF-100A USAF No52-5754
GEAR UP
SYM ALTITUDE AVG WEIGHT
FEET LBS
2300 23000
40000 23000
40000 23000
25000 23000
TOWER FLY BYS
AIRPLANE FLY BYS
PACE POINTS
RADAR TRACKED DIVE
NOSE BOOM AIRSPEED SYSTEM
139.25"
TOP VIEW
SIDE VIEW
4.25
TOTAL PRESSURE PICK-UP
2.6" O.D. TUBE
STATIC SOURCE
3 HOLES-.1935" DIA.
30"
ΔV CORRECTION TO BE ADDED IN KNOTS
14
12
10
8
6
4
2
0
-2
0.3
0.4
0.5
0.6
0.7
0.8
0.9
1.0
1.1
1.2
1.3
1.4
INDICATED MACH NUMBER

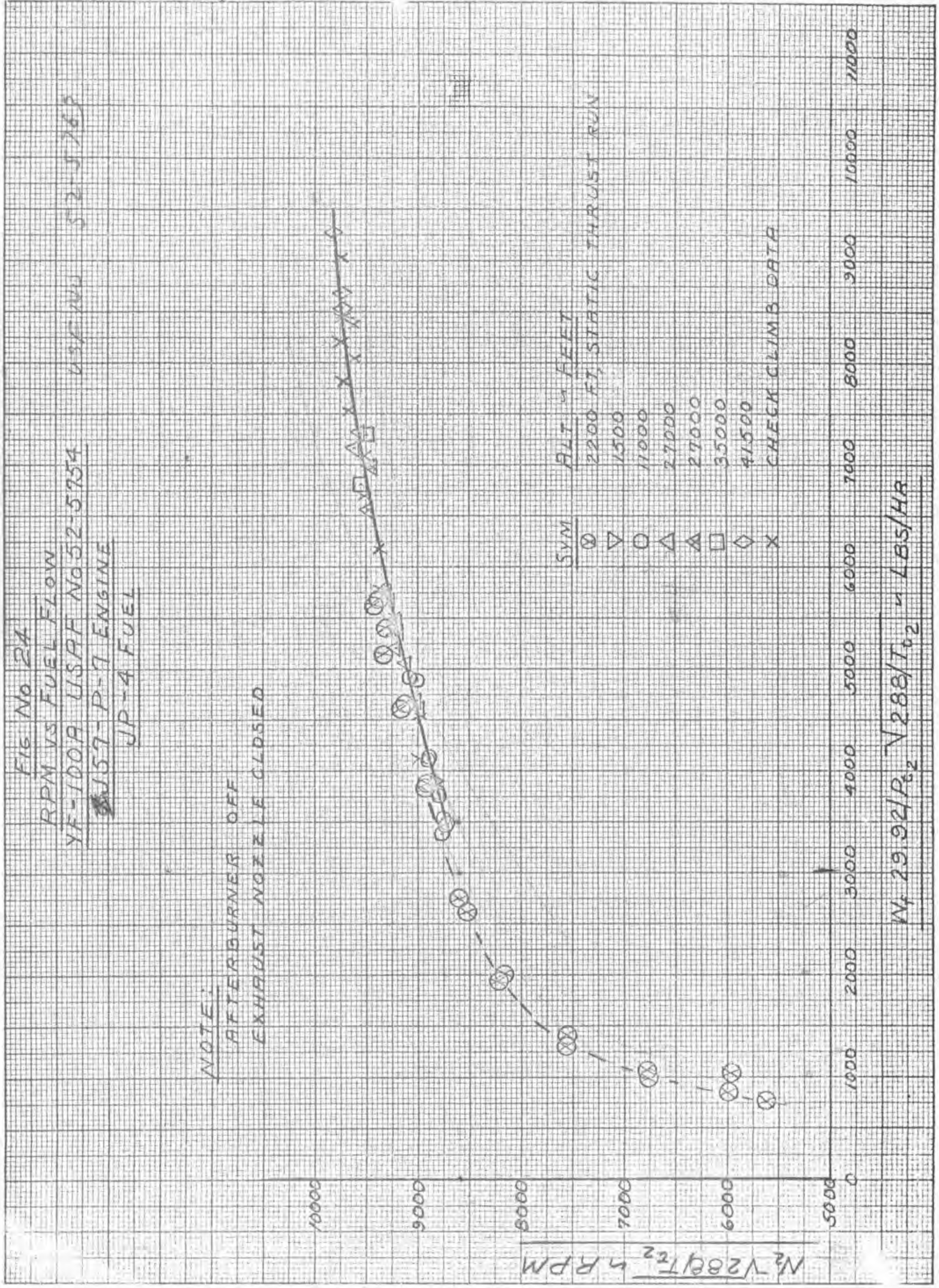


FIG No 24
RPM vs FUEL FLOW
YF-100A USAF No 52-5754
J57-P-7 ENGINE
JP-4 FUEL
NOTE:
AFTERBURNER OFF
EXHAUST NOZZLE CLOSED
SYM ALT ~ FEET
2200 FT, STATIC THRUST RUN
1500
11000
27000
27000
35000
41500
CHECK CLIMB DATA
0
1000
2000
3000
4000
5000
6000
7000
8000
9000
10000
11000
5000
6000
7000
8000
9000
10000
$W_f\ 29.92/P_{t2}\ \sqrt{288/T_{t2}}$ ~ LBS/HR
$N\sqrt{288/T_{t2}}$ ~ RPM

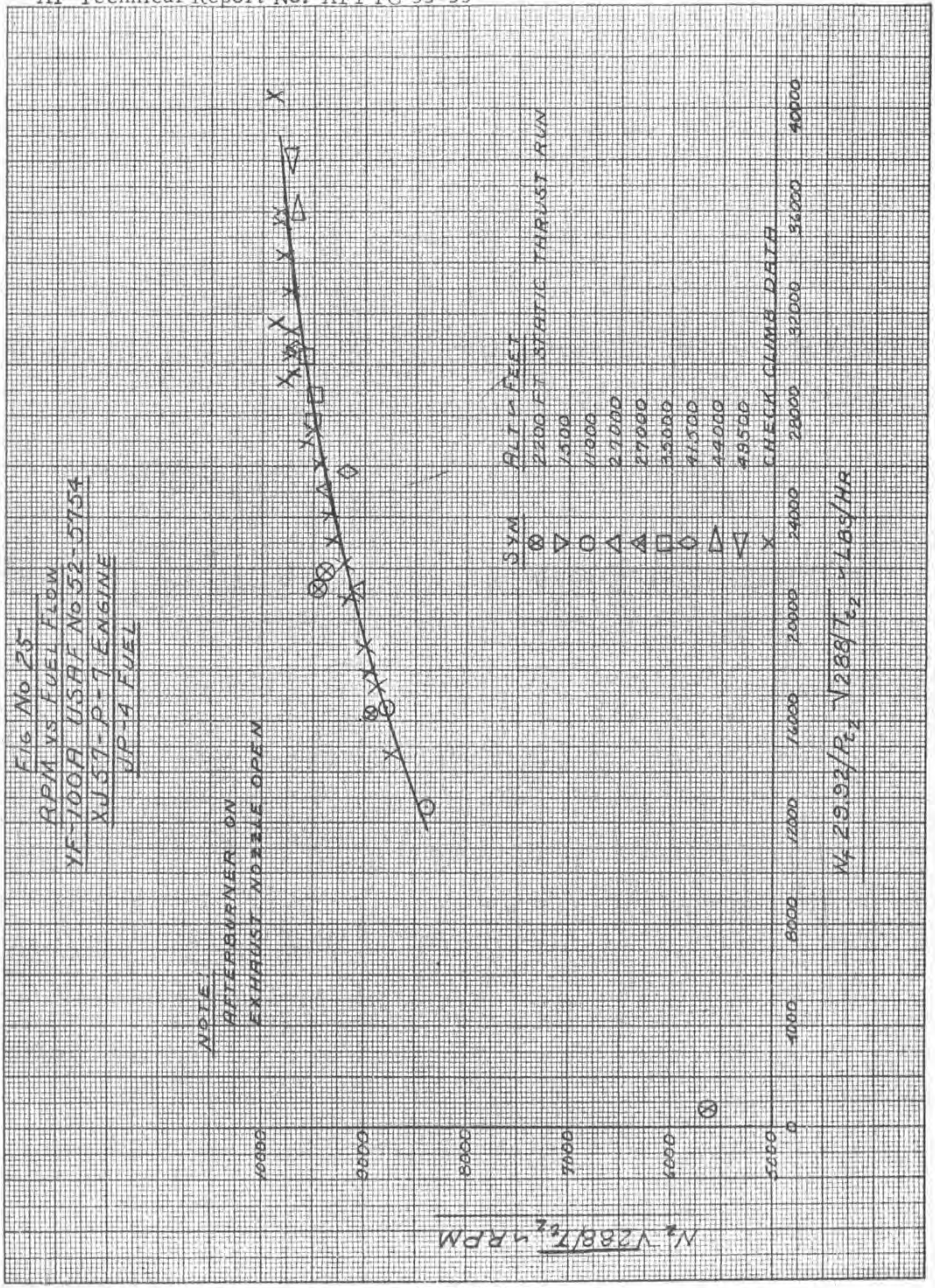


Fig No 25
RPM vs FUEL FLOW
YF-100A USAF No 52-5754
XJ57-P-7 ENGINE
JP-4 FUEL
NOTE:
AFTERBURNER ON
EXHAUST NOZZLE OPEN
SYM
ALT ~ FEET
2300 FT STATIC THRUST RUN
1500
11000
21000
27000
35000
41500
44000
49500
CHECK CLIMB DATA
N_f 29.92/P_{t_2} $\sqrt{288/T_{t_2}}$ ~ LBS/HR
N_2 $\sqrt{288/T_{t_2}}$ ~ RPM
0
4000
8000
12000
16000
20000
24000
28000
32000
36000
40000
5000
6000
7000
8000
9000
10000

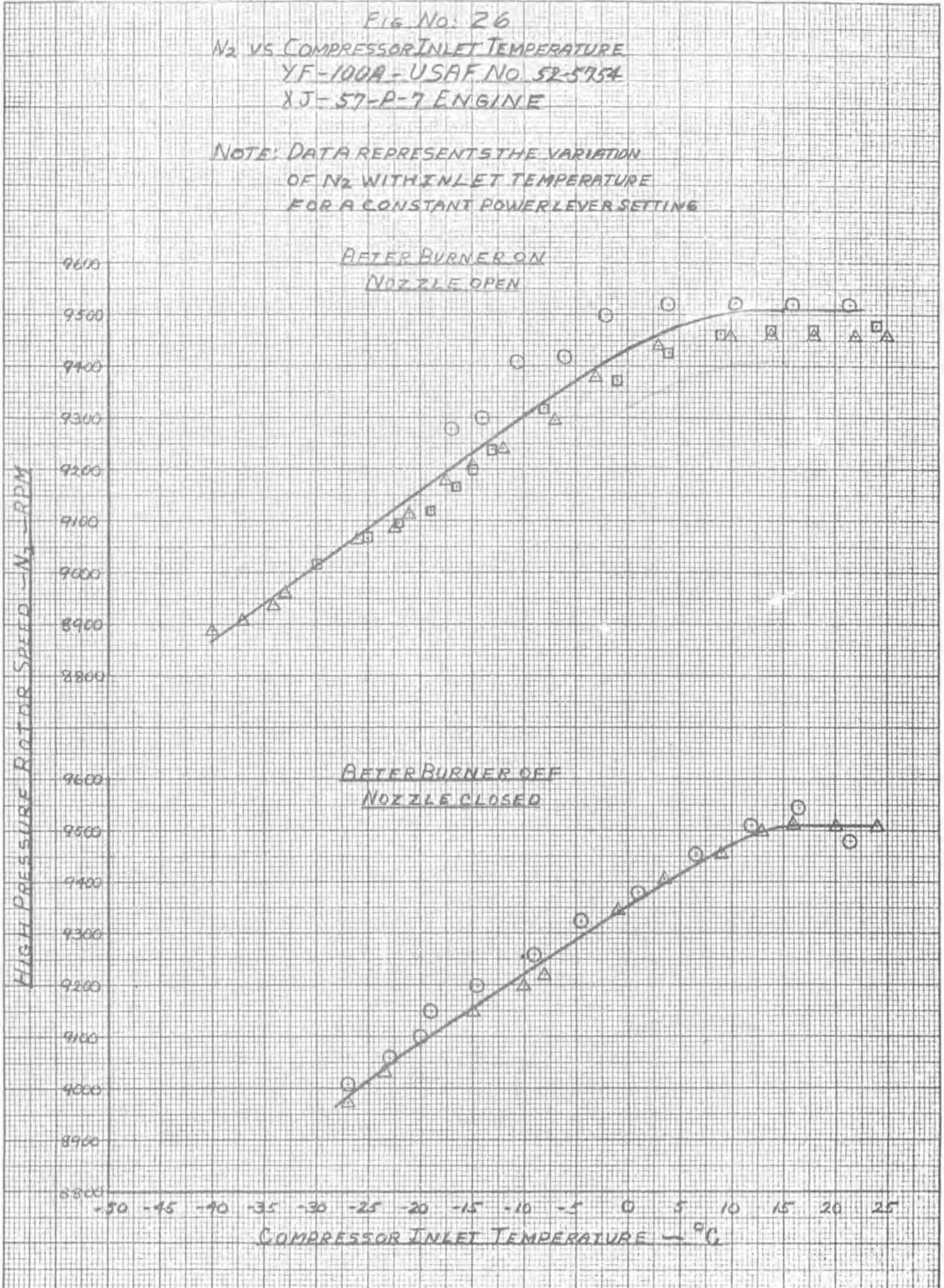


FIG NO: 26
N2 VS COMPRESSOR INLET TEMPERATURE
YF-100A - USAF NO 52-5754
XJ-57-P-7 ENGINE
NOTE: DATA REPRESENTS THE VARIATION
OF N2 WITH INLET TEMPERATURE
FOR A CONSTANT POWERLEVER SETTING
AFTER BURNER ON
NOZZLE OPEN
AFTER BURNER OFF
NOZZLE CLOSED
HIGH PRESSURE ROTOR SPEED - N2 - RPM
9600
9500
9400
9300
9200
9100
9000
8900
8800
-50 -45 -40 -35 -30 -25 -20 -15 -10 -5 0 5 10 15 20 25
COMPRESSOR INLET TEMPERATURE - °C

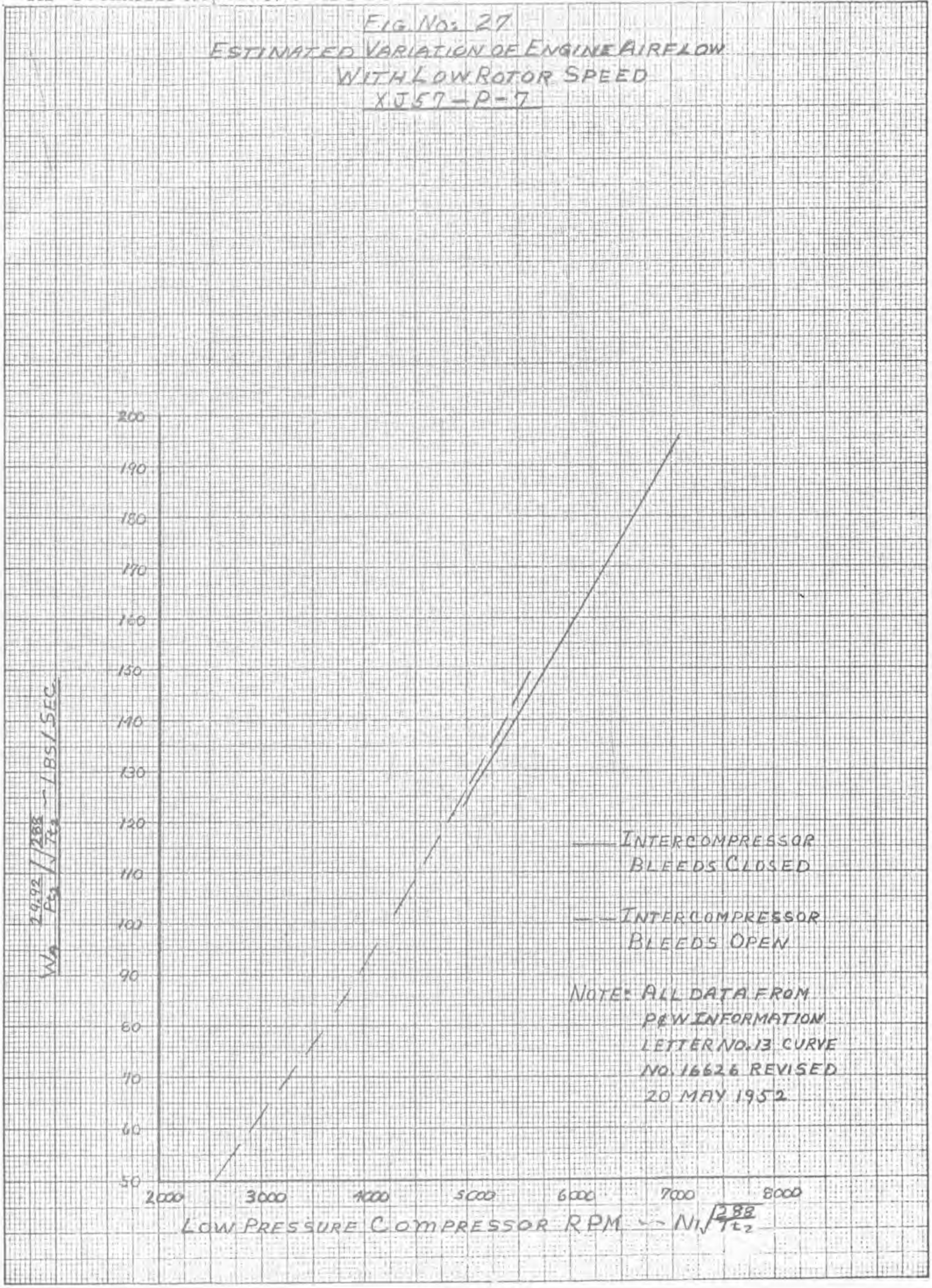


FIG. NO. 27
ESTIMATED VARIATION OF ENGINE AIRFLOW
WITH LOW ROTOR SPEED
XJ57-P-7
200
190
180
170
160
150
140
130
120
110
100
90
80
70
60
50
2000
3000
4000
5000
6000
7000
8000
LOW PRESSURE COMPRESSOR RPM ~ $N_1\sqrt{\frac{288}{T_{t2}}}$
$W_a \frac{29.92}{P_{t2}}\sqrt{\frac{288}{T_{t2}}}$ ~ LBS/SEC
— INTERCOMPRESSOR BLEEDS CLOSED
— — INTERCOMPRESSOR BLEEDS OPEN
NOTE: ALL DATA FROM P&W INFORMATION LETTER NO. 13 CURVE NO. 16626 REVISED 20 MAY 1952

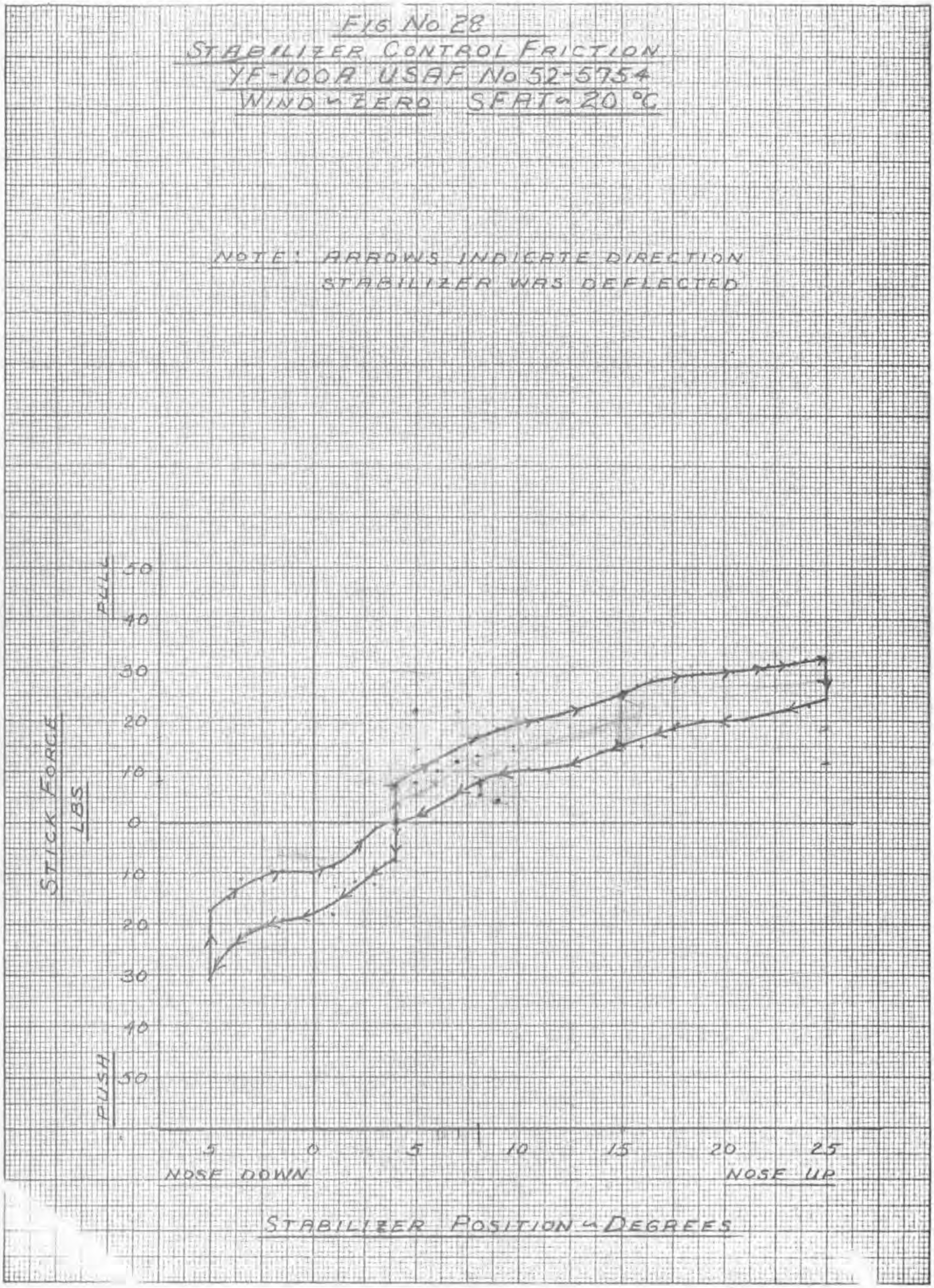


FIG No 28
STABILIZER CONTROL FRICTION
YF-100A USAF No 52-5754
WIND ~ ZERO SFAT ~ 20 °C
NOTE: ARROWS INDICATE DIRECTION
STABILIZER WAS DEFLECTED
STICK FORCE
LBS
PULL
PUSH
50
40
30
20
10
0
10
20
30
40
50
5
0
5
10
15
20
25
NOSE DOWN
NOSE UP
STABILIZER POSITION ~ DEGREES

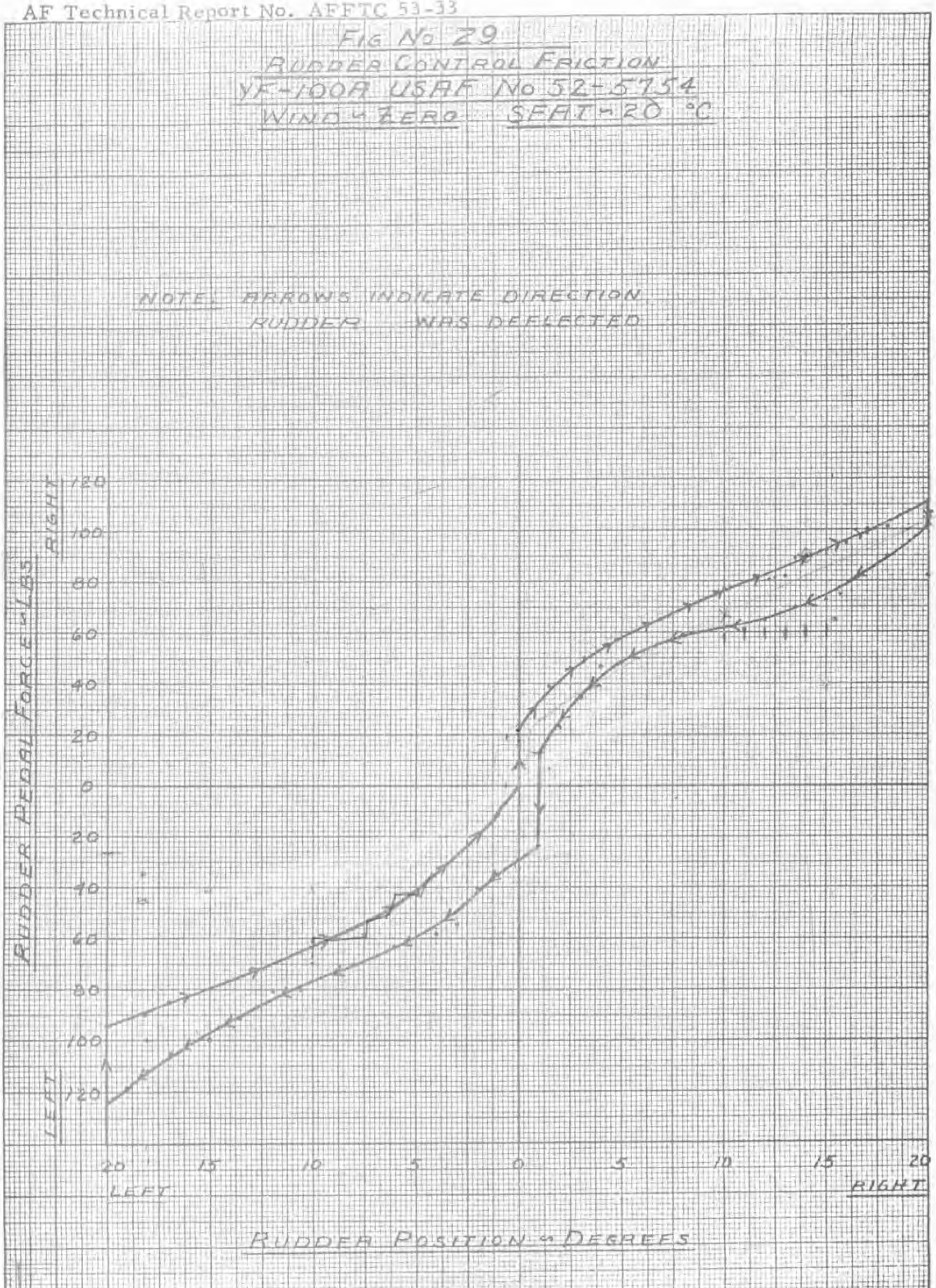


FIG NO 29
RUDDER CONTROL FRICTION
YF-100A USAF No 52-5754
WIND ~ ZERO SFAT ~ 20 °C
NOTE: ARROWS INDICATE DIRECTION RUDDER WAS DEFLECTED
RUDDER PEDAL FORCE ~ LBS
RIGHT
LEFT
120
100
80
60
40
20
0
20
40
60
80
100
120
20
15
10
5
0
5
10
15
20
LEFT
RIGHT
RUDDER POSITION ~ DEGREES

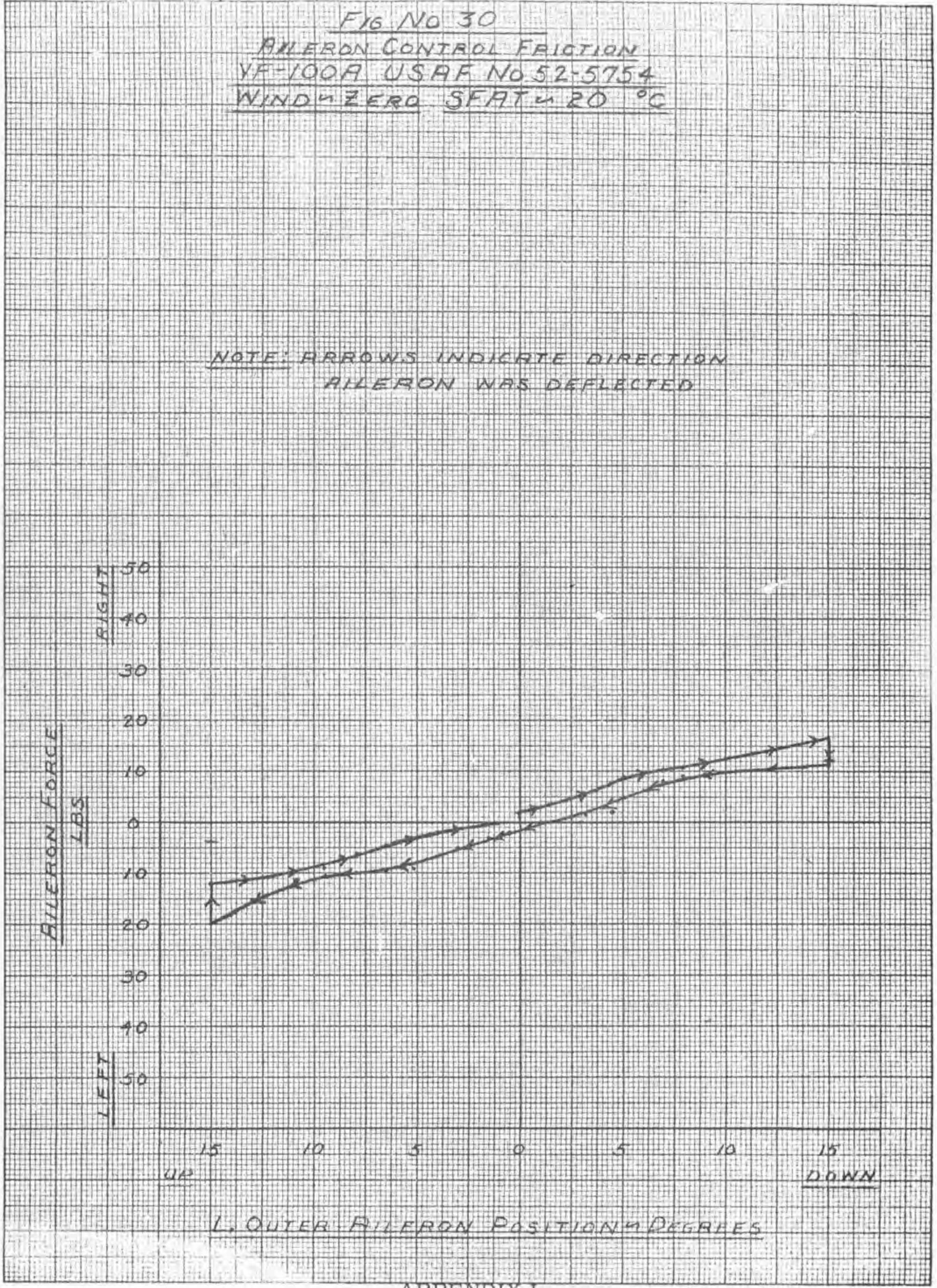


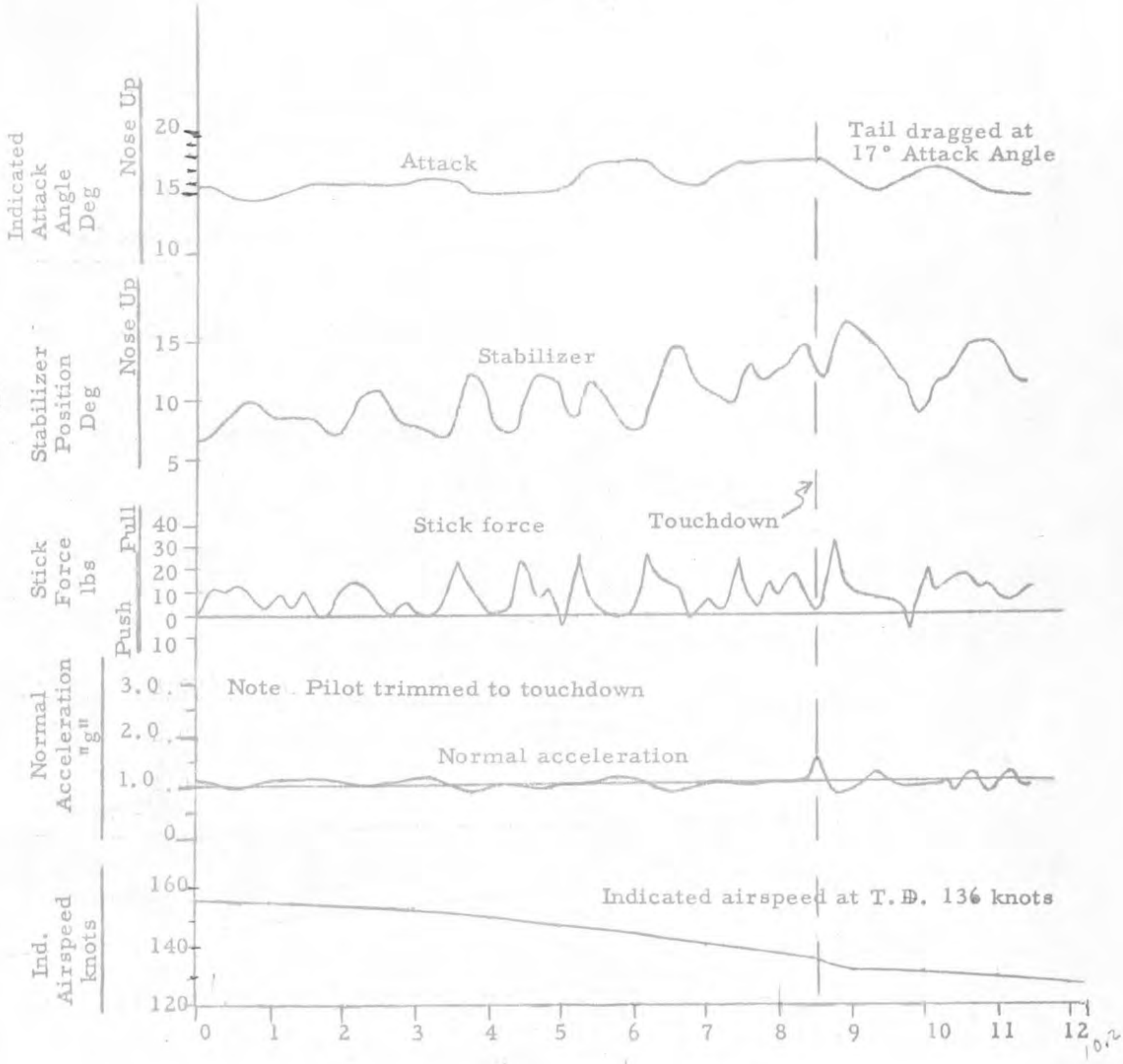
FIG NO 30
AILERON CONTROL FRICTION
YF-100A USAF No 52-5754
WIND ~ ZERO SFAT ~ 20 °C
NOTE: ARROWS INDICATE DIRECTION
AILERON WAS DEFLECTED
AILERON FORCE
LBS
RIGHT
LEFT
50
40
30
20
10
0
10
20
30
40
50
15
10
5
0
5
10
15
UP
DOWN
L. OUTER AILERON POSITION ~ DEGREES

Figure No. 31

LANDING TIME HISTORY
YF-100A, USAF No. 52-5754
Landing Configuration
TRIM CONDITIONS

Altitude 2,250 feet Weight 21,800 lbs
CG 29.5 $^{o}/o$ MAC Dive Brake Full Open

No external stores attached



Time - Seconds

APPENDIX I

Figure No. 32

STALL TIME HISTORY

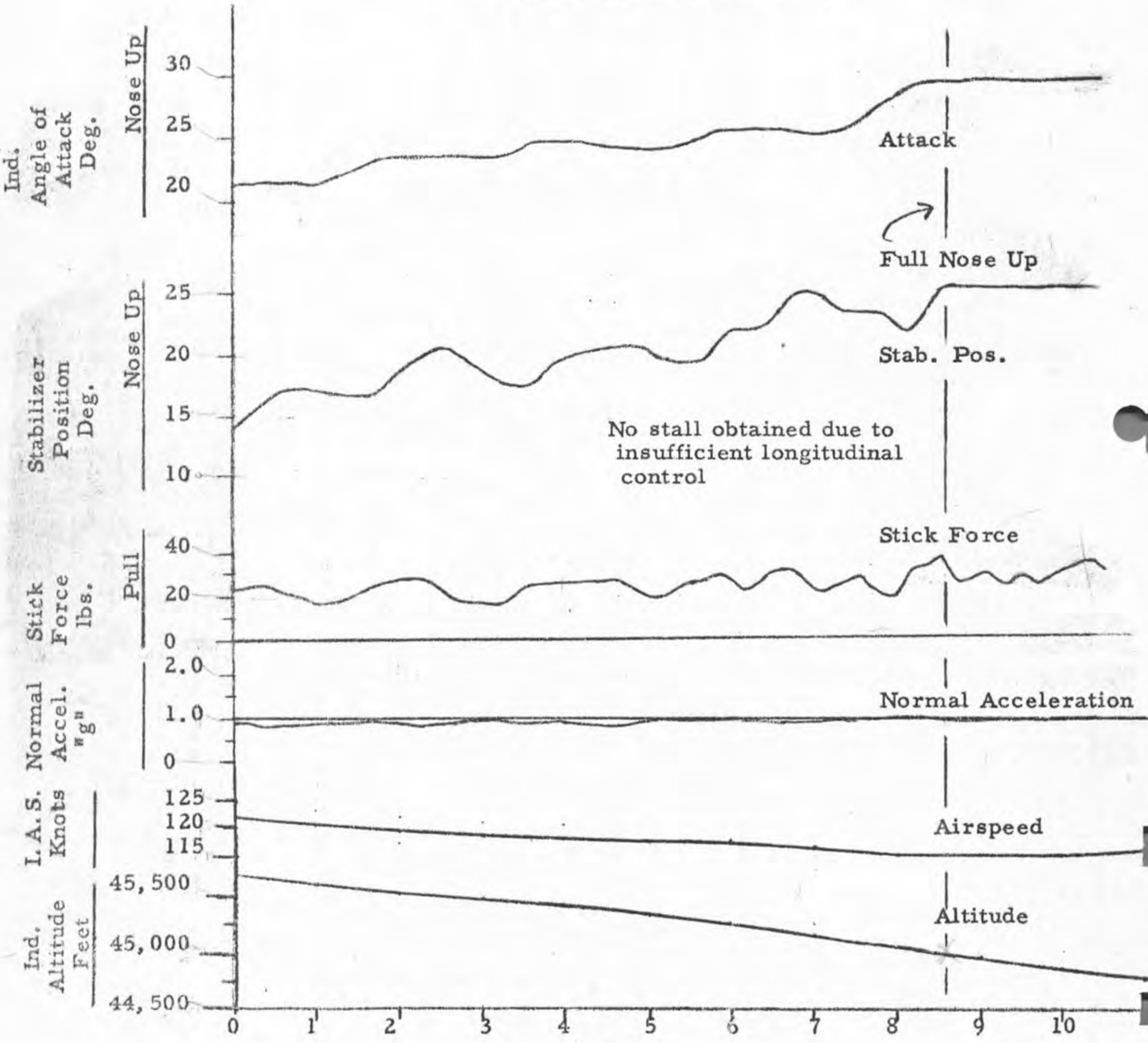
YF-100A, USAF No. 52-5754
Power Configuration (A/B off)

TRIM CONDITIONS

CAS 220 knots
CG 30 % mac
N_2 8950 rpm

Altitude 44,800 feet
Weight 21,500 lbs
Stab. Pos. 4.3° nose up

No external stores attached



APPENDIX I

Figure No. 32

STALL TIME HISTORY

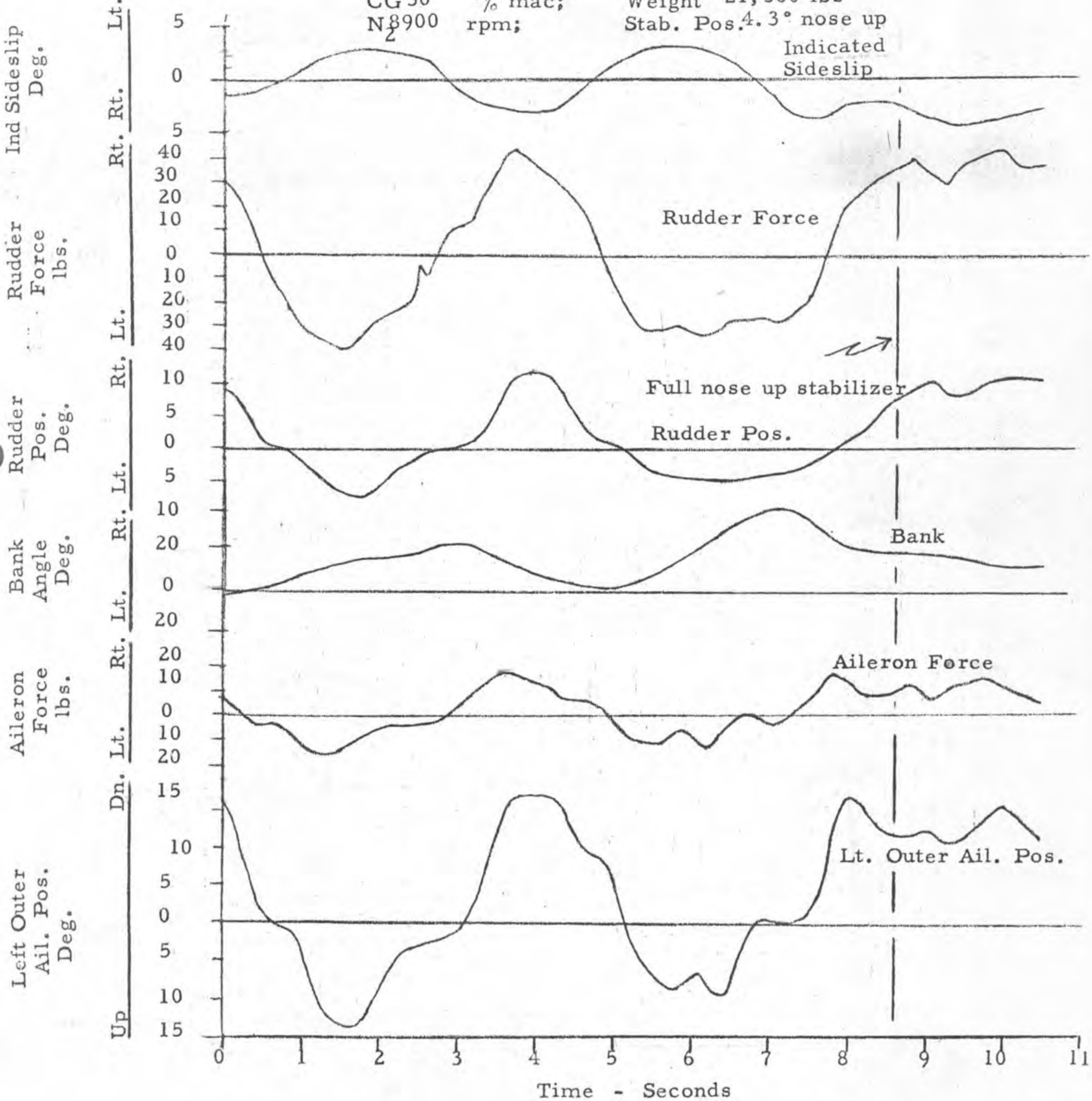
YF-100A, USAF No. 52-5754

Power Configuration (A/B off)

TRIM CONDITIONS

CAS 220 knots; CG 30 °/o mac; N_2 8900 rpm;

Altitude 44,800 feet; Weight 21,500 lbs; Stab. Pos. 4.3° nose up



APPENDIX I

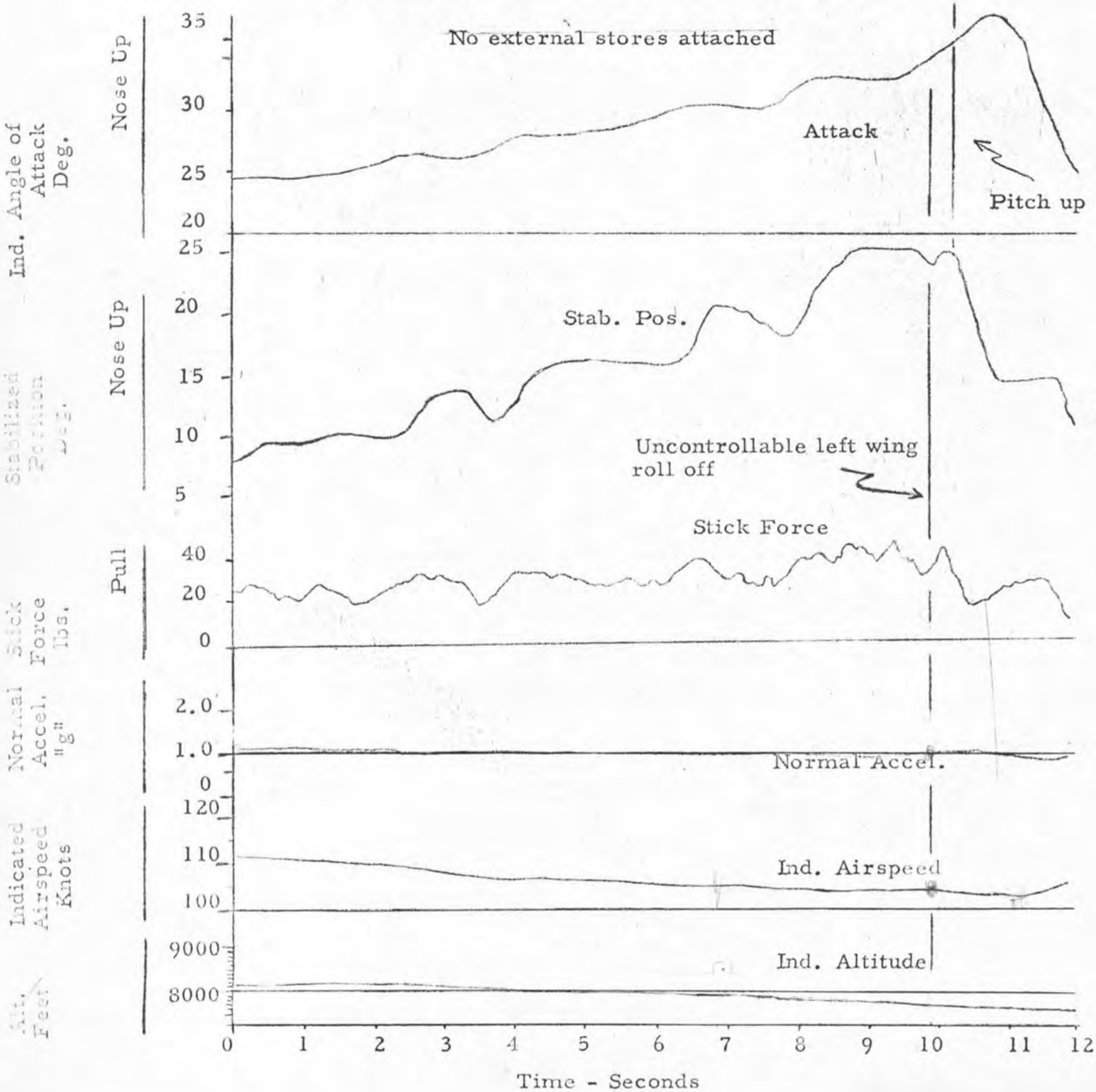
AF Technical report No. AFFTC 53-33

Figure No. 33

STALL TIME HISTORY
YF 100A, USAF No. 52-5754
Power Approach Configuration

TRIM CONDITIONS

IASl. 55.5 knots	Altitude 8040 feet
CG 30.6 °/₀ mac;	Weight 21,100 lbs
N_2 9100 rpm;	Stab. Pos 7.8 nose up



APPENDIX I

Figure No. 33

STALL TIME HISTORY

YF 100 A, USAF No. 52-5754

Power Approach Configuration

TRIM CONDITIONS

IAS 155.5 knots; CG 30.6 %mac; N_2 9100 rpm; Altitude 8040 feet; Weight 21000 lbs; Stab. Pos. 7.8° nose up

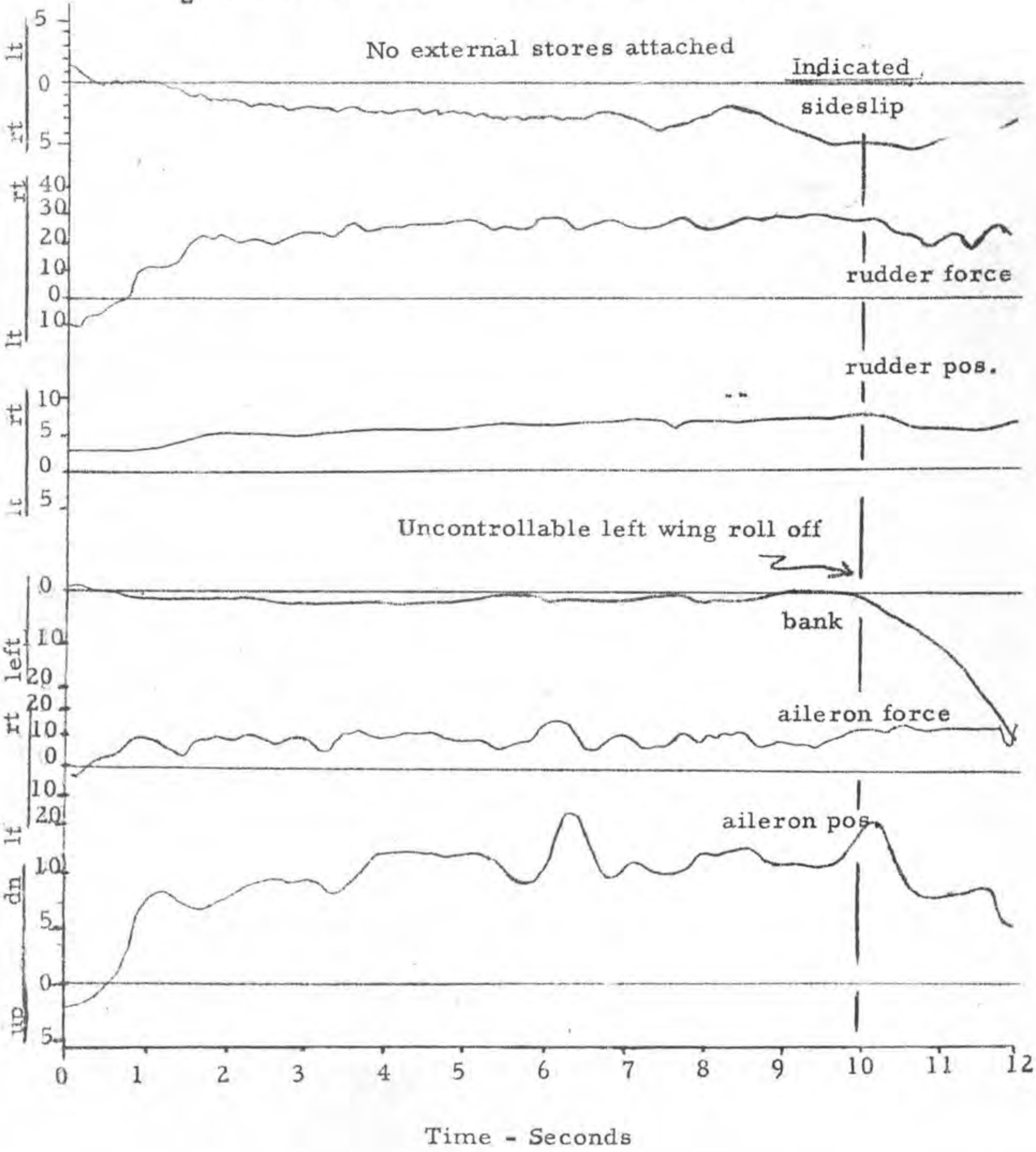


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Figure No. 34

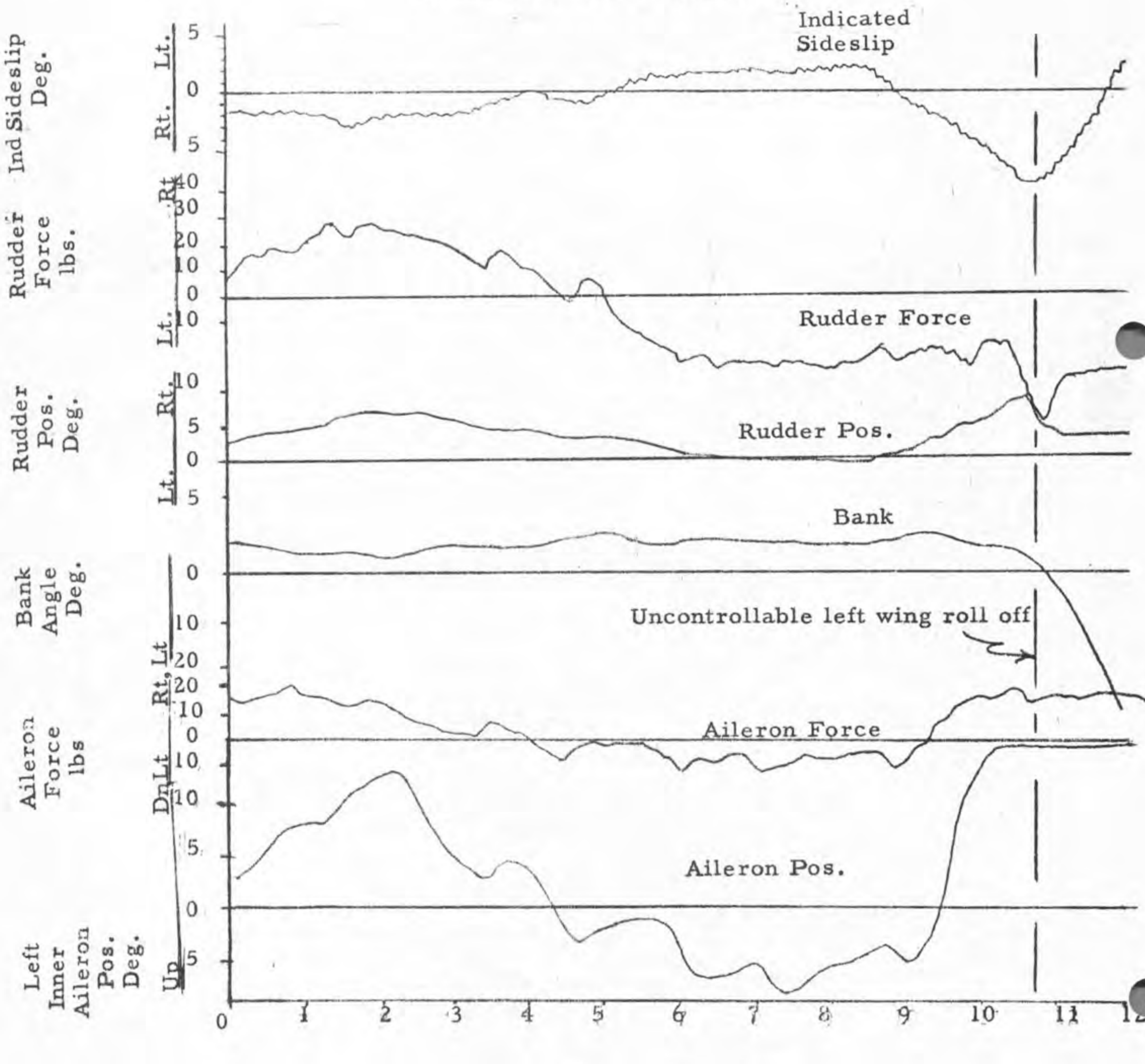
STALL TIME HISTORY

YF 100 A, USAF No. 52-5754

Power Approach Configuration

CAS 157.5 knots	Altitude 10,340 feet
CG 29.9 % mac;	Weight 21,250 lbs
N_2 9100 rpm;	Stab. Pos. 7.7° nose up

No external stores attached



Time - Seconds

APPENDIX I

Figure No. 3 4

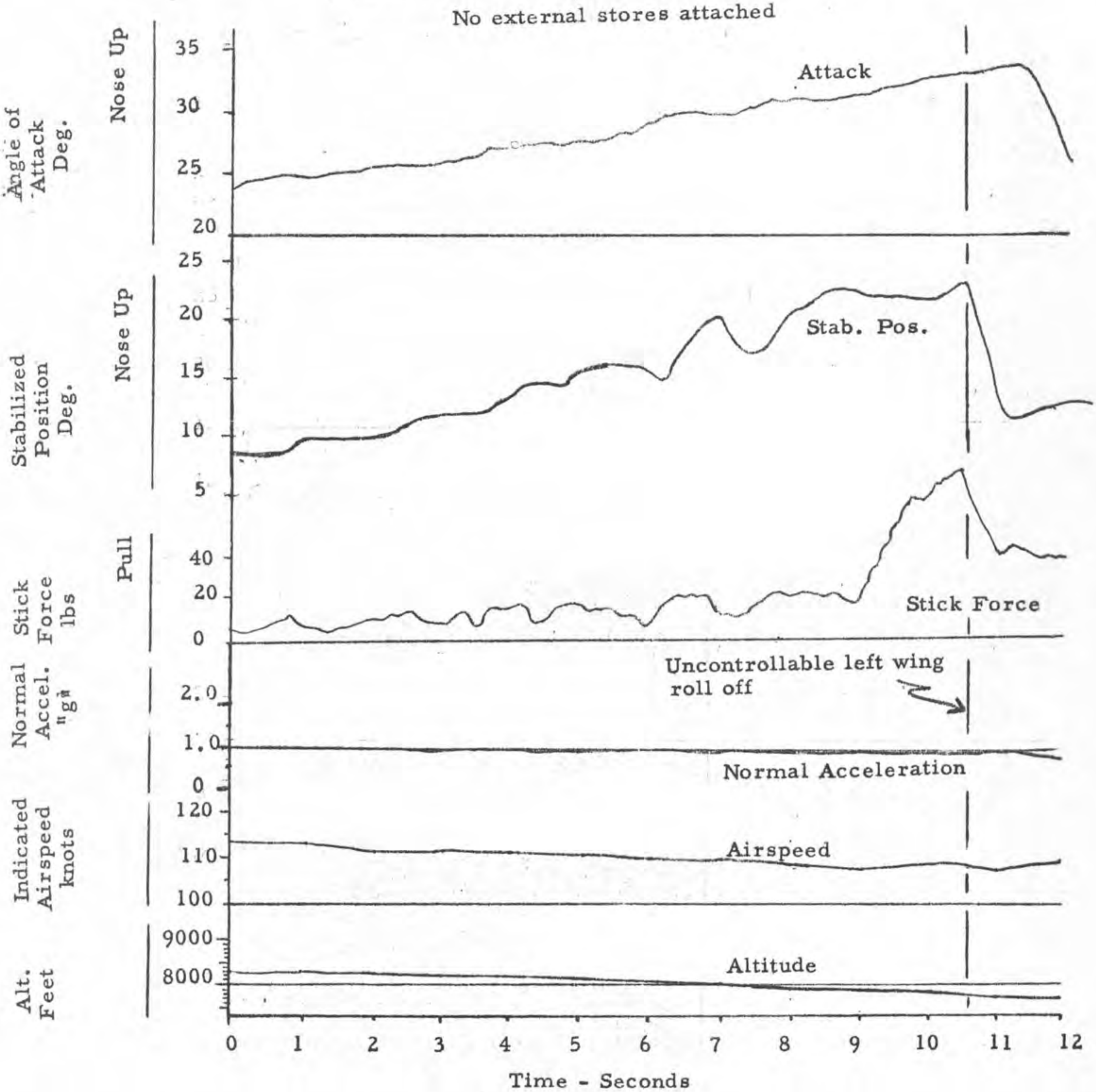
STALL TIME HISTORY
YF 100A, USAF No. 52-5754
Powered Approach Configuration

TRIM CONDITIONS

CAS 157.5 knots
CG 29.9% mac;
N_2 9100rpm;

Altitude 10,340 feet
Weight 21,250 lbs
Stab. Pos. 7.7° nose up

No external stores attached



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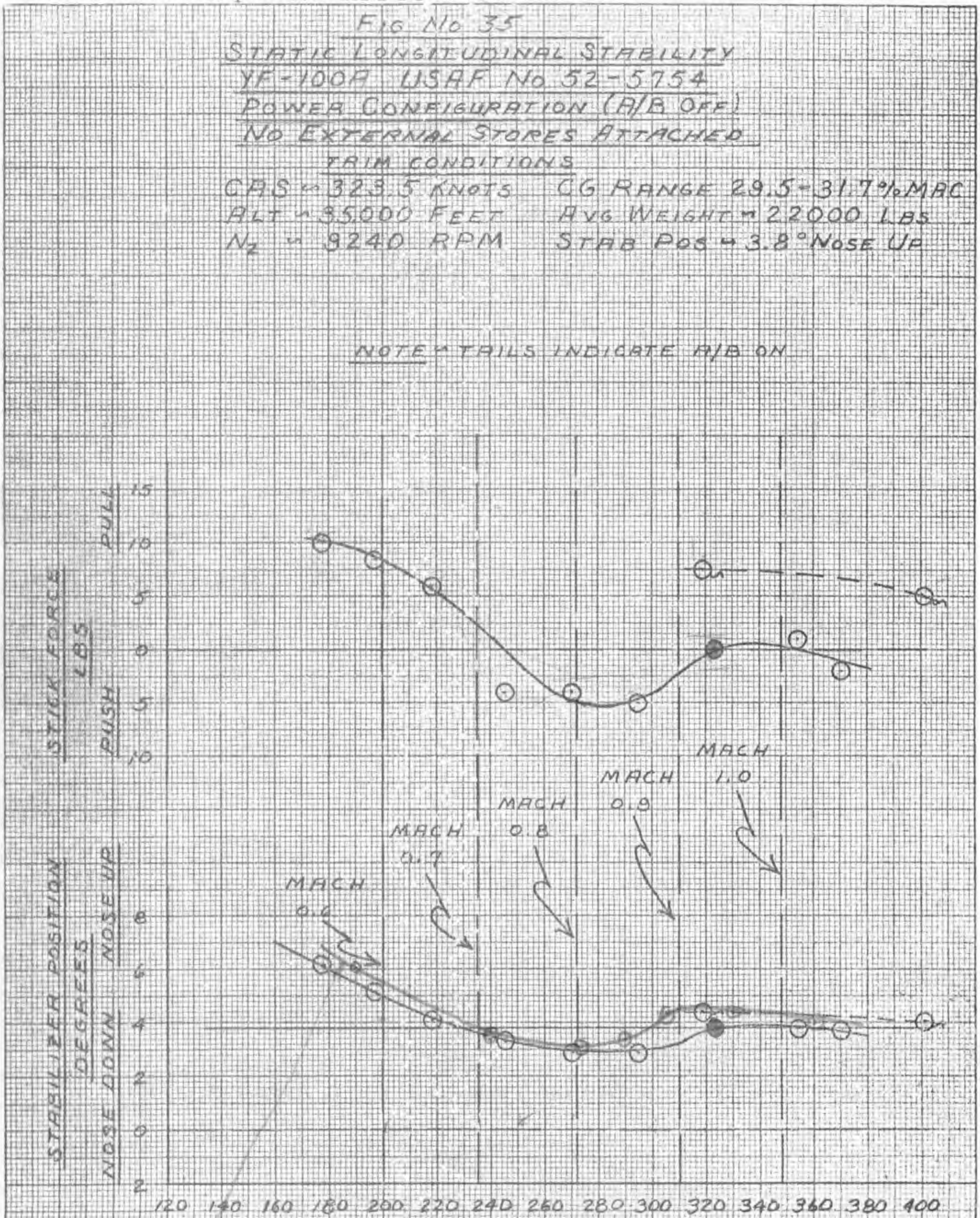


FIG NO 35
STATIC LONGITUDINAL STABILITY
YF-100A USAF NO 52-5754
POWER CONFIGURATION (A/B OFF)
NO EXTERNAL STORES ATTACHED
TRIM CONDITIONS
CAS ~ 323.5 KNOTS
ALT ~ 35000 FEET
N_2 ~ 9240 RPM
CG RANGE 29.5 - 31.7% MAC
AVG WEIGHT ~ 22000 LBS
STAB POS ~ 3.8° NOSE UP
NOTE ~ TAILS INDICATE A/B ON
STICK FORCE LBS
PUSH
PULL
15
10
5
0
5
10
STABILIZER POSITION DEGREES
NOSE DOWN
NOSE UP
8
6
4
2
0
2
MACH 0.6
MACH 0.7
MACH 0.8
MACH 0.9
MACH 1.0
120 140 160 180 200 220 240 260 280 300 320 340 360 380 400
CALIBRATED AIRSPEED ~ KNOTS

40,000' - fwd.

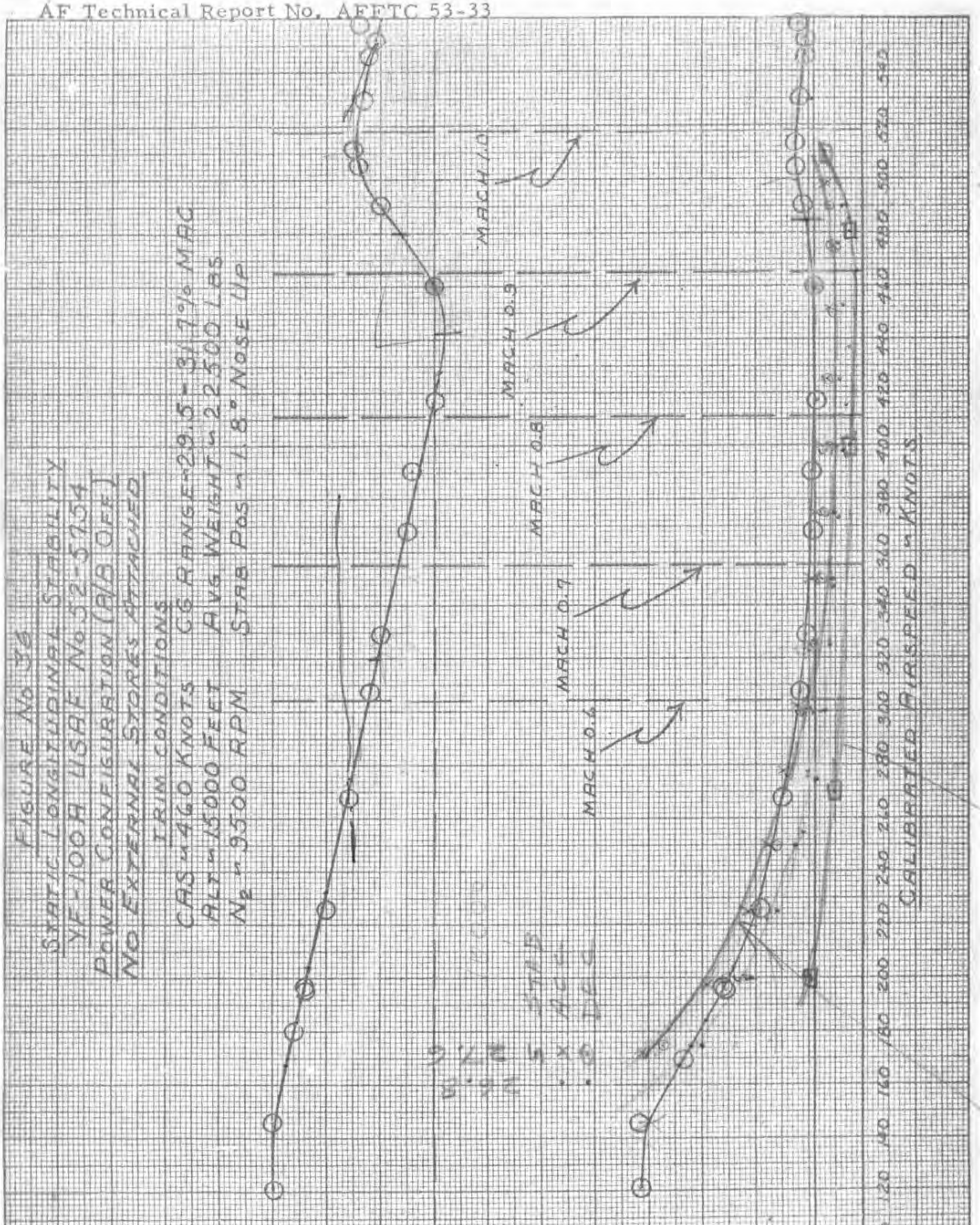


FIGURE No 33
STATIC LONGITUDINAL STABILITY
YF-100A USAF No 52-5754
POWER CONFIGURATION (A/B OFF)
NO EXTERNAL STORES ATTACHED
TRIM CONDITIONS
CAS ≈ 460 KNOTS CG RANGE ≈ 29.5 - 31.7% MAC
ALT ≈ 15000 FEET AVG WEIGHT ≈ 22500 LBS
N2 ≈ 9500 RPM STAB POS ≈ 1.8° NOSE UP
MACH 0.6
MACH 0.7
MACH 0.8
MACH 0.9
MACH 1.0
STICK FORCE
PUSH LBS PULL
STABILIZER POSITION
DEG NOSE UP
CALIBRATED AIRSPEED ~ KNOTS

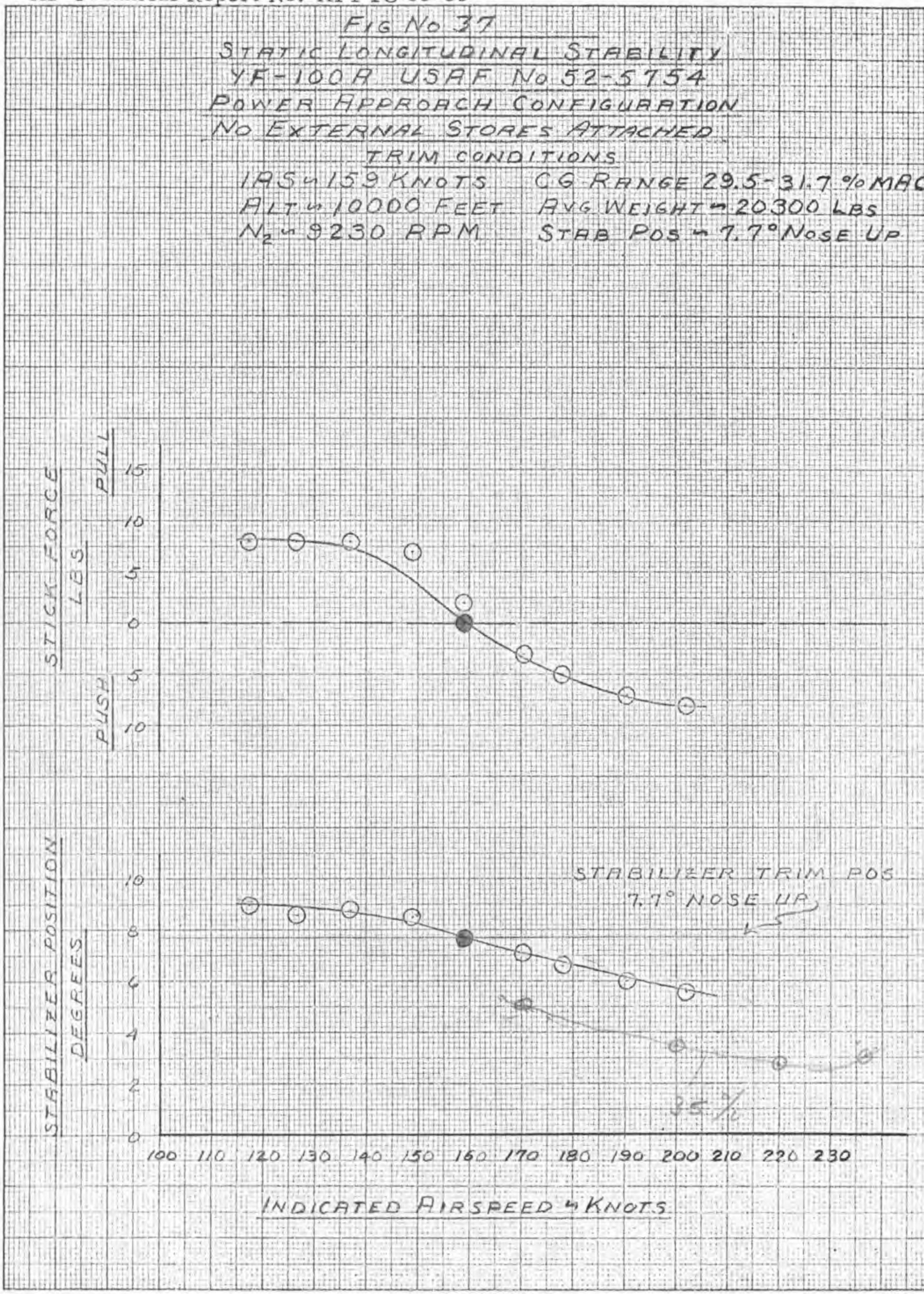


FIG NO 37
STATIC LONGITUDINAL STABILITY
YF-100A USAF No 52-5754
POWER APPROACH CONFIGURATION
NO EXTERNAL STORES ATTACHED
TRIM CONDITIONS
IAS ~ 159 KNOTS
CG RANGE 29.5-31.7 % MAC
ALT ~ 10000 FEET
AVG WEIGHT ~ 20300 LBS
N2 ~ 9230 RPM
STAB POS ~ 7.7° NOSE UP
STICK FORCE LBS
PULL
PUSH
15
10
5
0
5
10
STABILIZER POSITION DEGREES
STABILIZER TRIM POS
7.7° NOSE UP
35%
10
8
6
4
2
0
100 110 120 130 140 150 160 170 180 190 200 210 220 230
INDICATED AIRSPEED ~ KNOTS

Figure No. 38

DYNAMIC LONGITUDINAL STABILITY
YF-100A, USAF No. 52-5754
Power Configuration (A/B On) Controls Free

TRIM CONDITIONS

CAS 274 knots
CG 31.0 $^o/_o$ MAC
Ave N_2 9070 RPM
Altitude 45,300 feet
Weight 22,800 lbs
Stabilizer 5.7° N.U.

No external stores attached

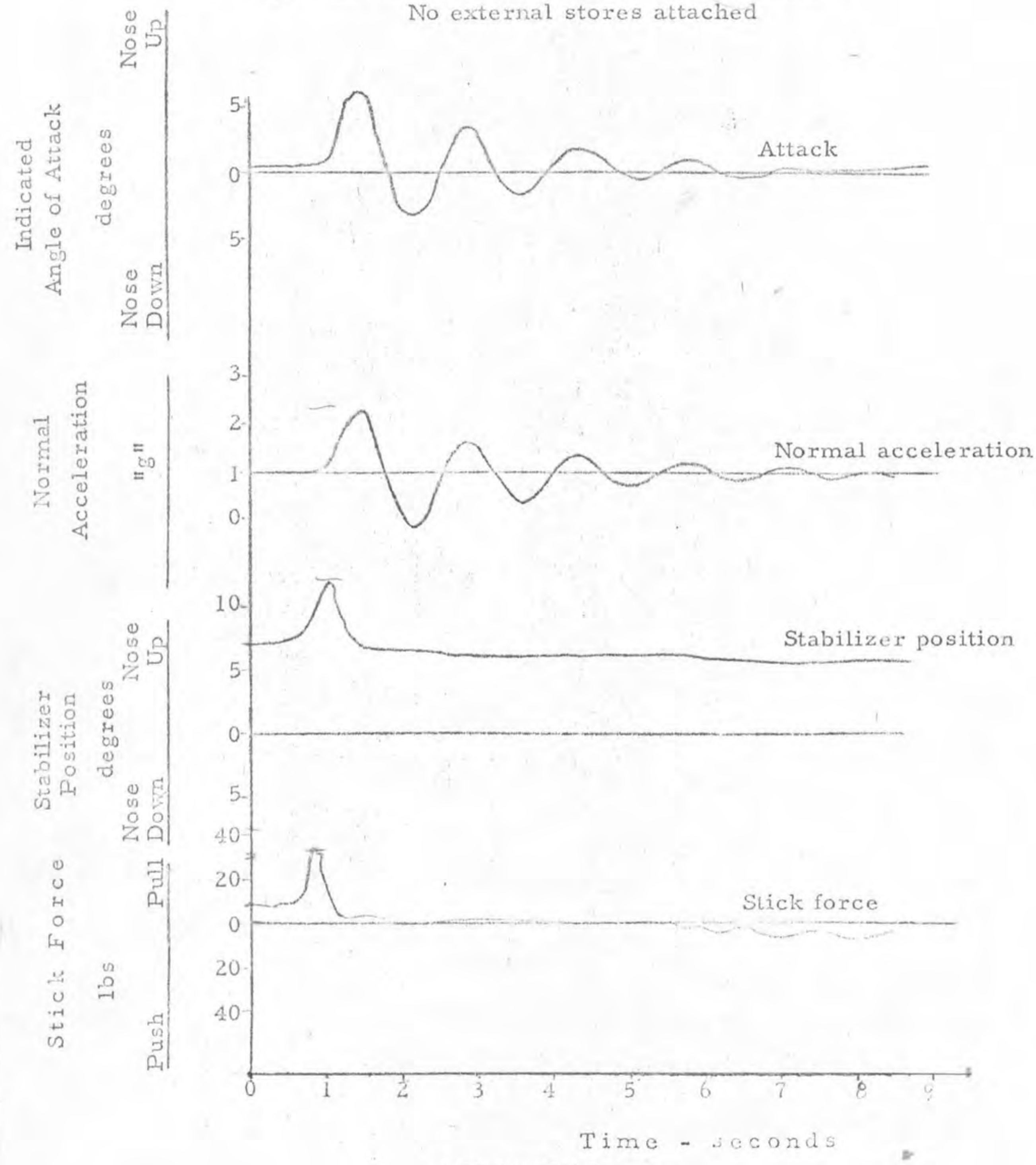


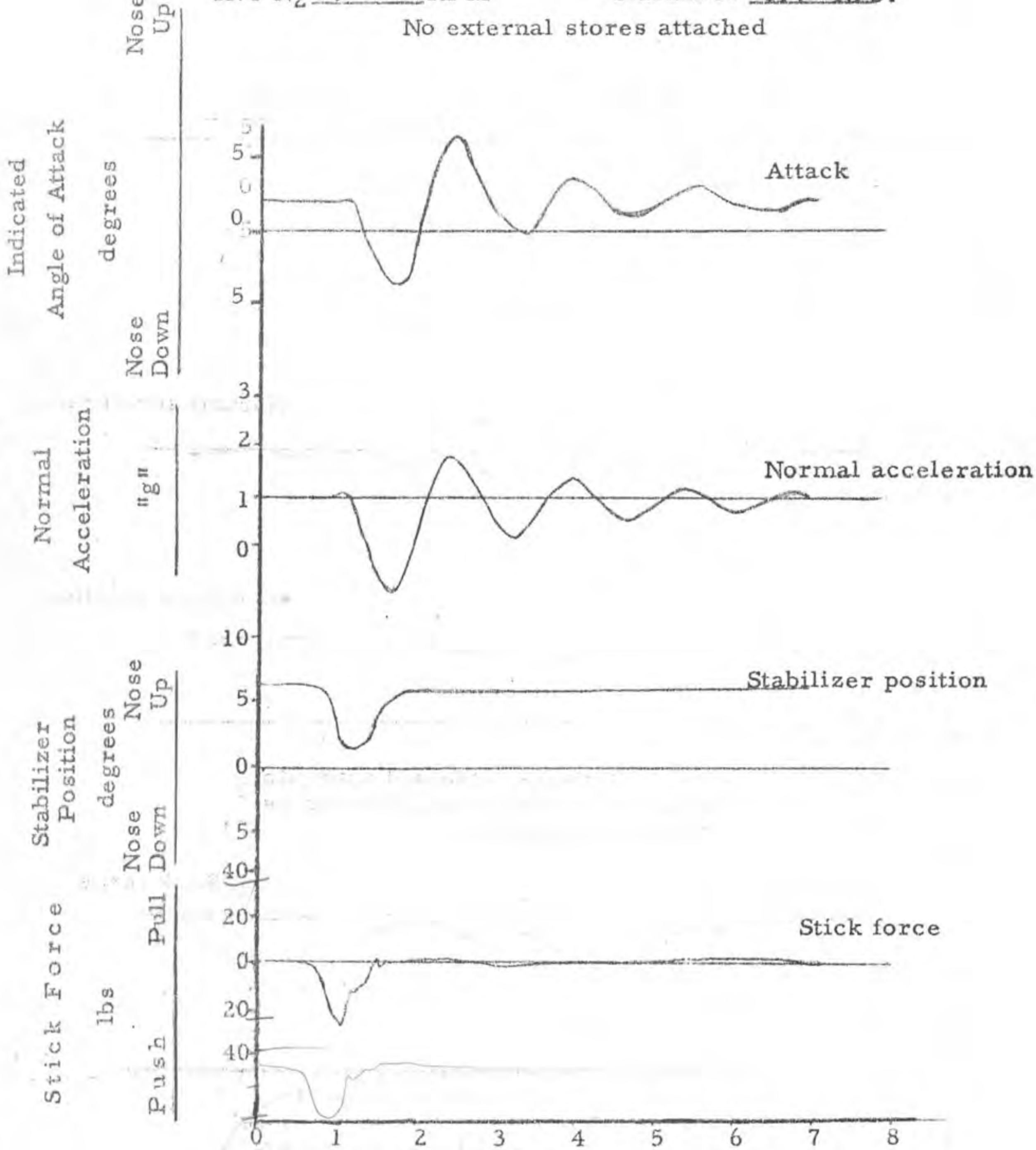
Figure No. 39

DYNAMIC LONGITUDINAL STABILITY
YF-100A, USAF No. 52-5754
Power Configuration (A/B On) Controls Free

TRIM CONDITIONS

CAS 274 knots Altitude 45,300 feet
CG 31.0 o/o MAC Weight 22,800 lbs
Ave N_2 9070 RPM Stabilizer 5.7° N.U.

No external stores attached



Time - seconds

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Figure No. 40

DYNAMIC LONGITUDINAL STABILITY

YF-100A, USAF No. 52-5754

Power Configuration (A/B on)

TRIM CONDITIONS

CAS 274 knots		Altitude 45,300 feet	
CG 31.0 $^{o}/o$ MAC		Weight 22,800 lbs	
Ave N_2 9070 RPM		Stabilizer 5.7° N.U.	

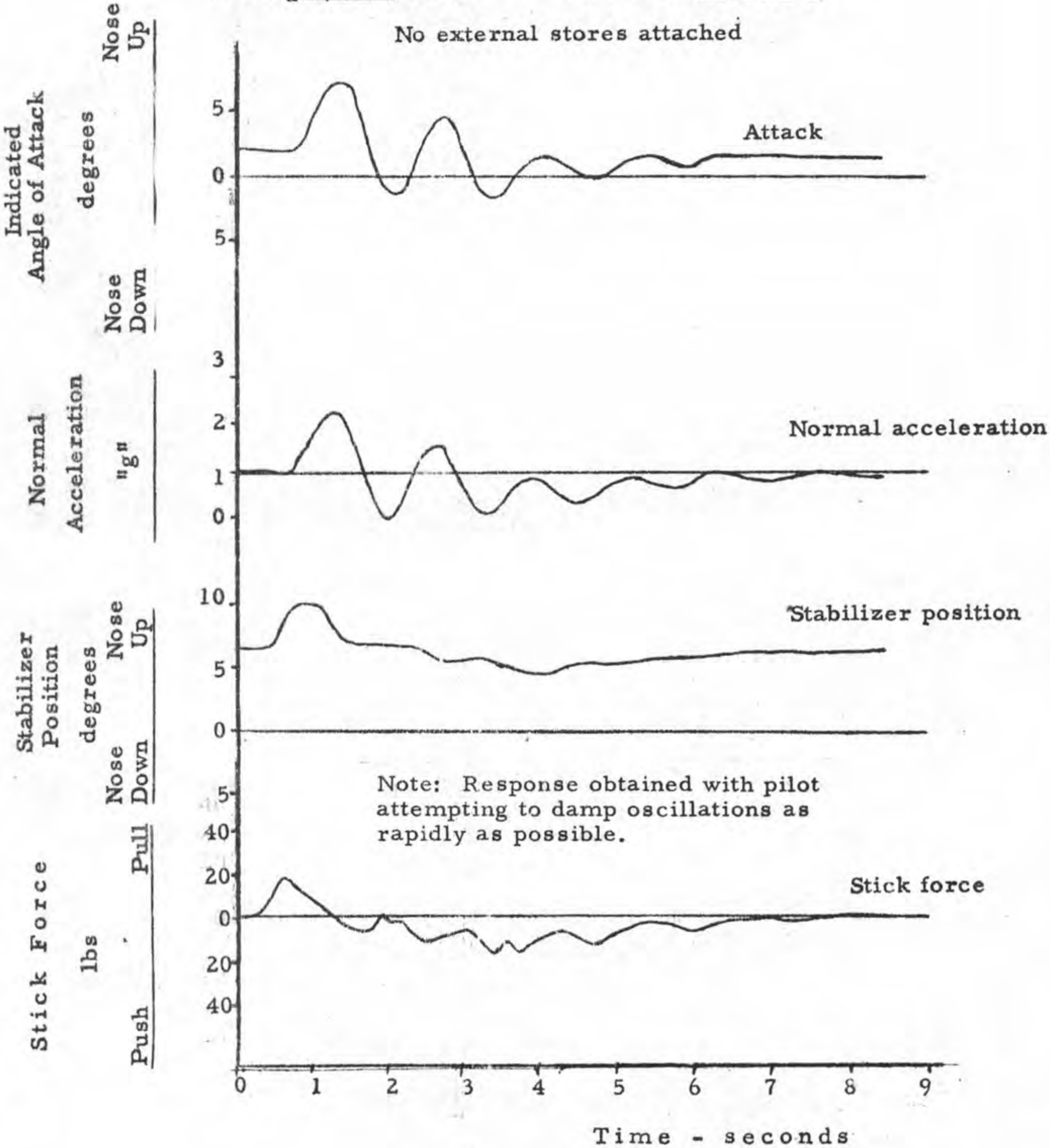

Figure No. 41

DYNAMIC LONGITUDINAL STABILITY
YF-100A, USAF No. 52-5754
Power Configuration (A/B on)

TRIM CONDITIONS

CAS 274 knots
CG 31.0 o/o MAC
Ave N_2 9070 RPM

Altitude 45,300 feet
Weight 22,800 lbs
Stabilizer 5.7° N.U.

No external stores attached

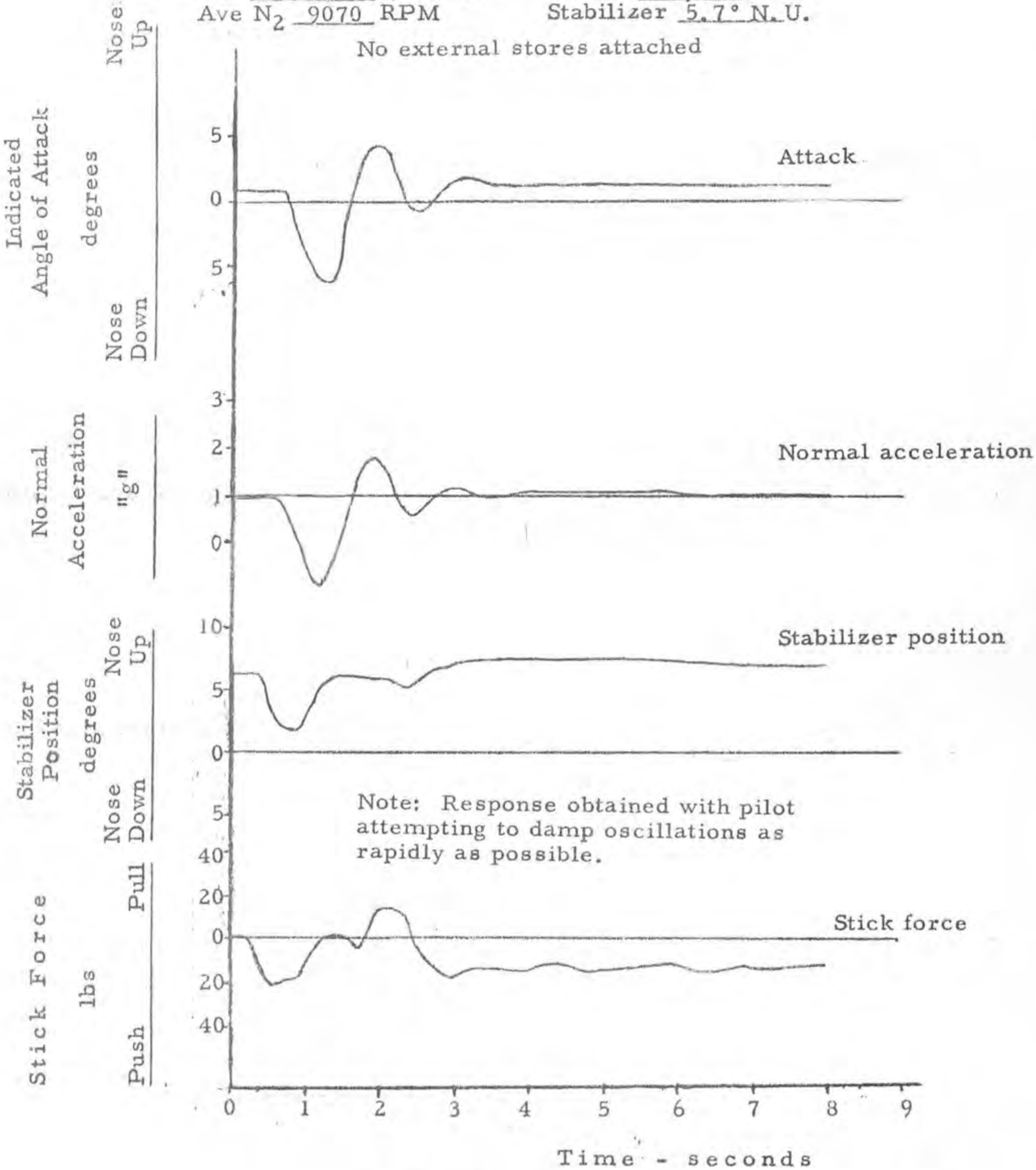

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Figure No. 42

DYNAMIC LONGITUDINAL STABILITY
YF-100A, USAF No. 52-5754
Power Configuration (A/B Off) Controls Free

TRIM CONDITIONS

CAS 475 knots
CG 31.2 o/o MAC
Ave N_2 9550 RPM
Altitude 10,500 feet
Weight 23,600 lbs
Stabilizer 1.7° N.U.

No external stores attached

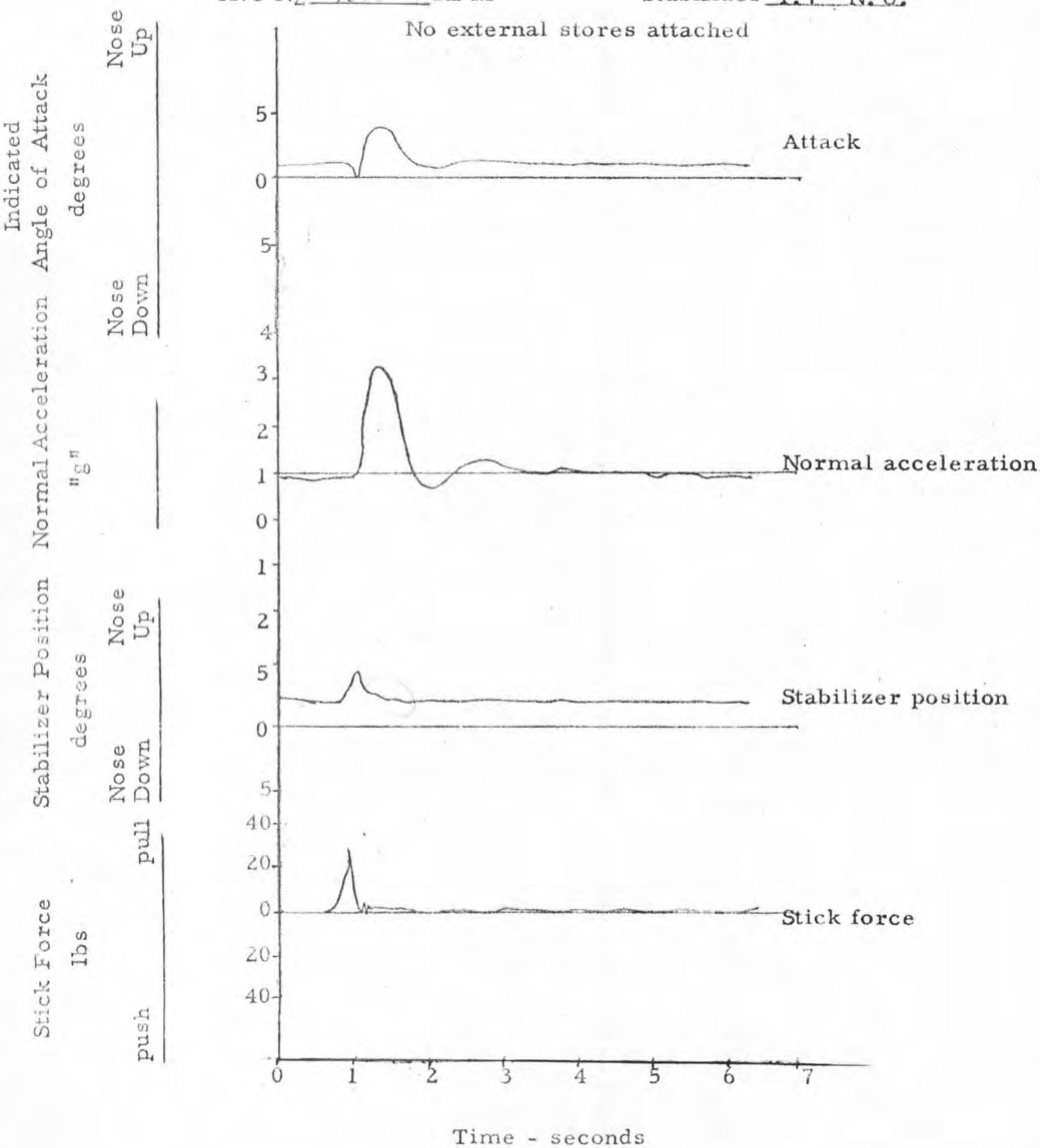

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Figure No. 43

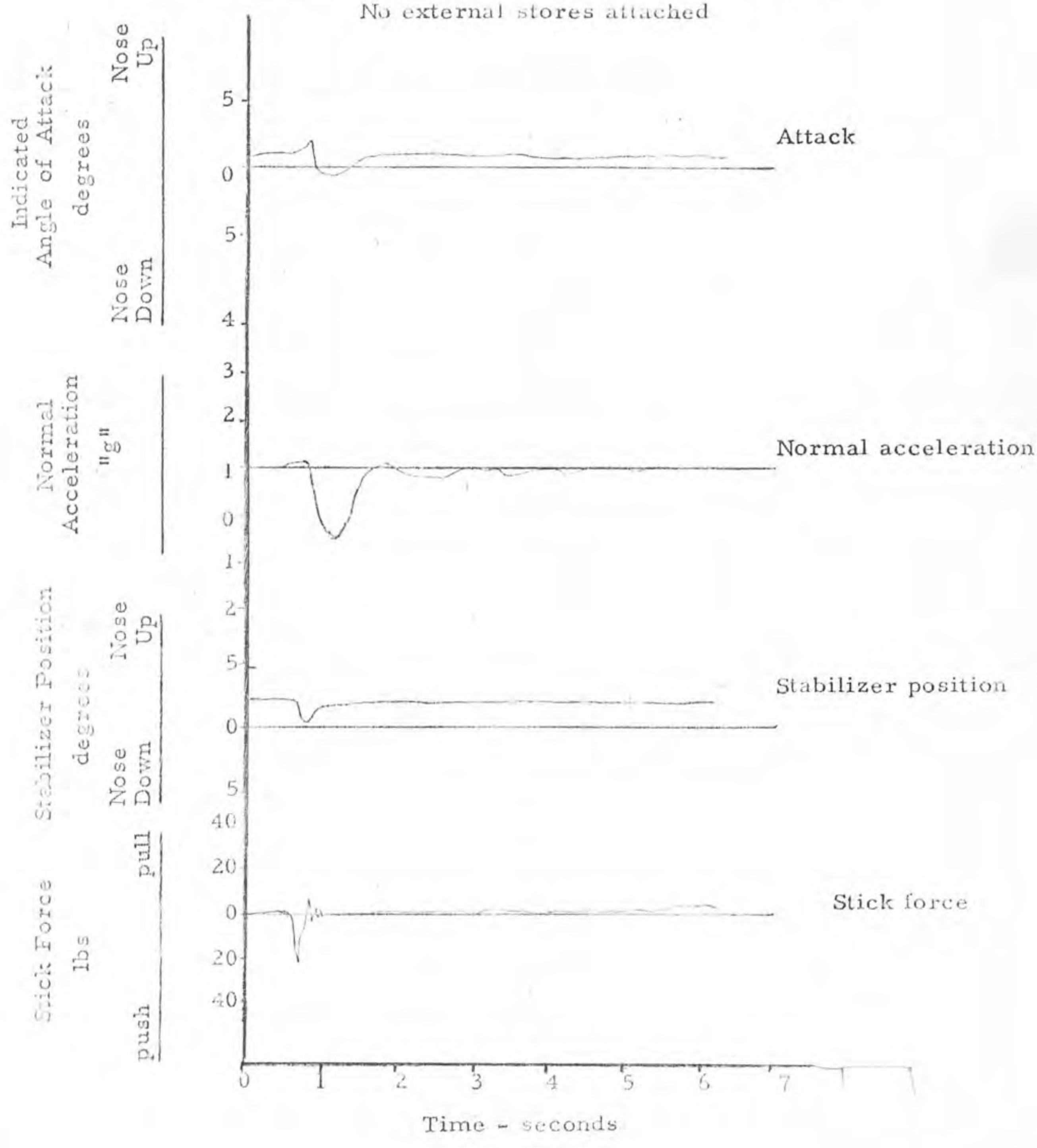
DYNAMIC LONGITUDINAL STABILITY
YF-100A, USAF No. 52-5754
Power Configuration (A/B Off) Controls Free

TRIM CONDITIONS

CAS 475 knots
CG 31.2 o/o MAC
Ave N_2 9550 RPM

Altitude 10,500 feet
Weight 23,600 lbs
Stabilizer 1.7° N.U.

No external stores attached



APPENDIX I

Figure No. 44

DYNAMIC LONGITUDINAL STABILITY
YF-100A, USAF No. 52-5754
Power Configuration (A/B off)

TRIM CONDITIONS

CAS 475 knots		Altitude 10,500 feet	
CG 31.2 °/o MAC		Weight 23,600 lbs	
Ave N_2 9550 RPM		Stabilizer 1.7° N.U.	

No external stores attached

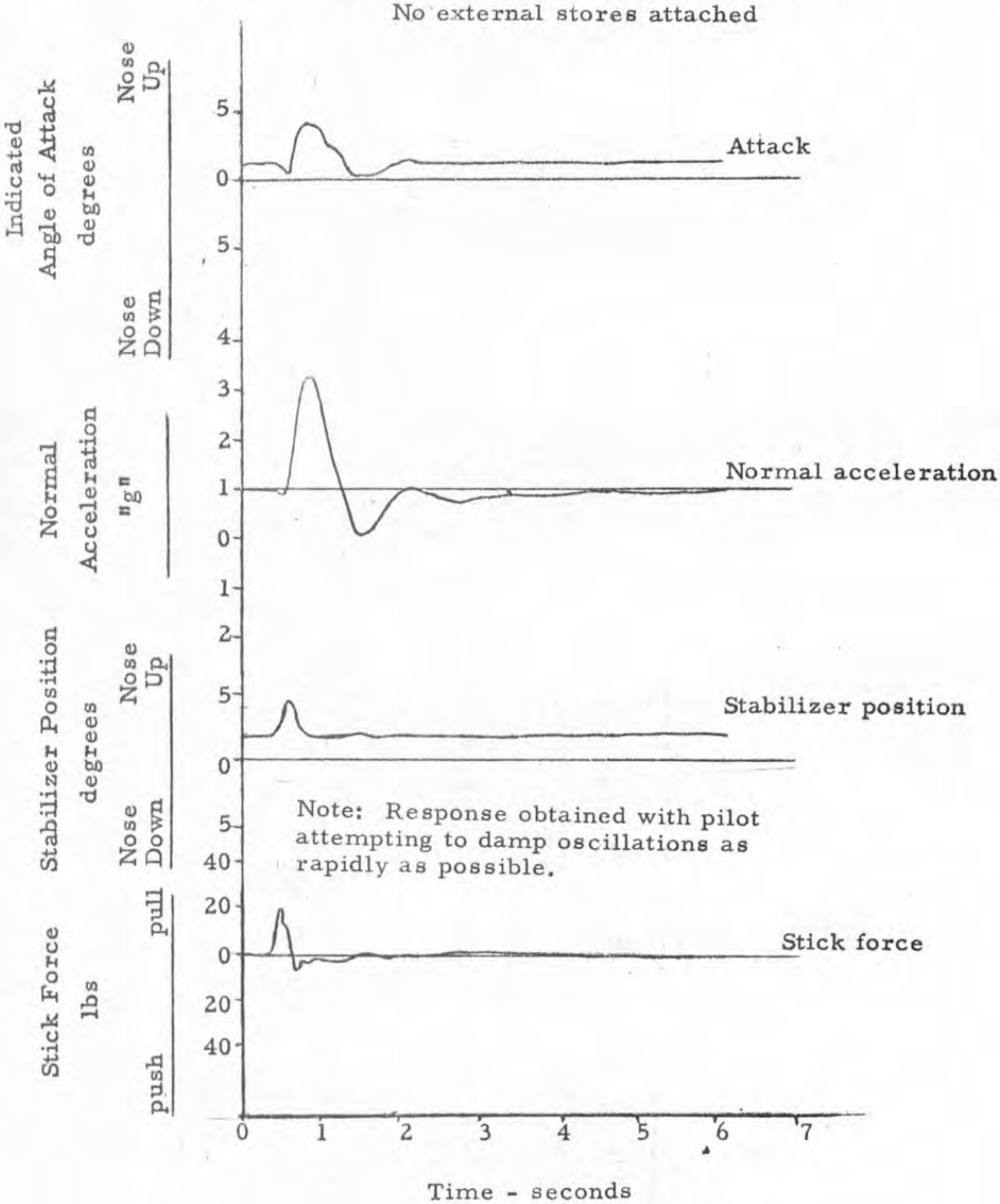


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Figure No. 45

DYNAMIC LONGITUDINAL STABILITY

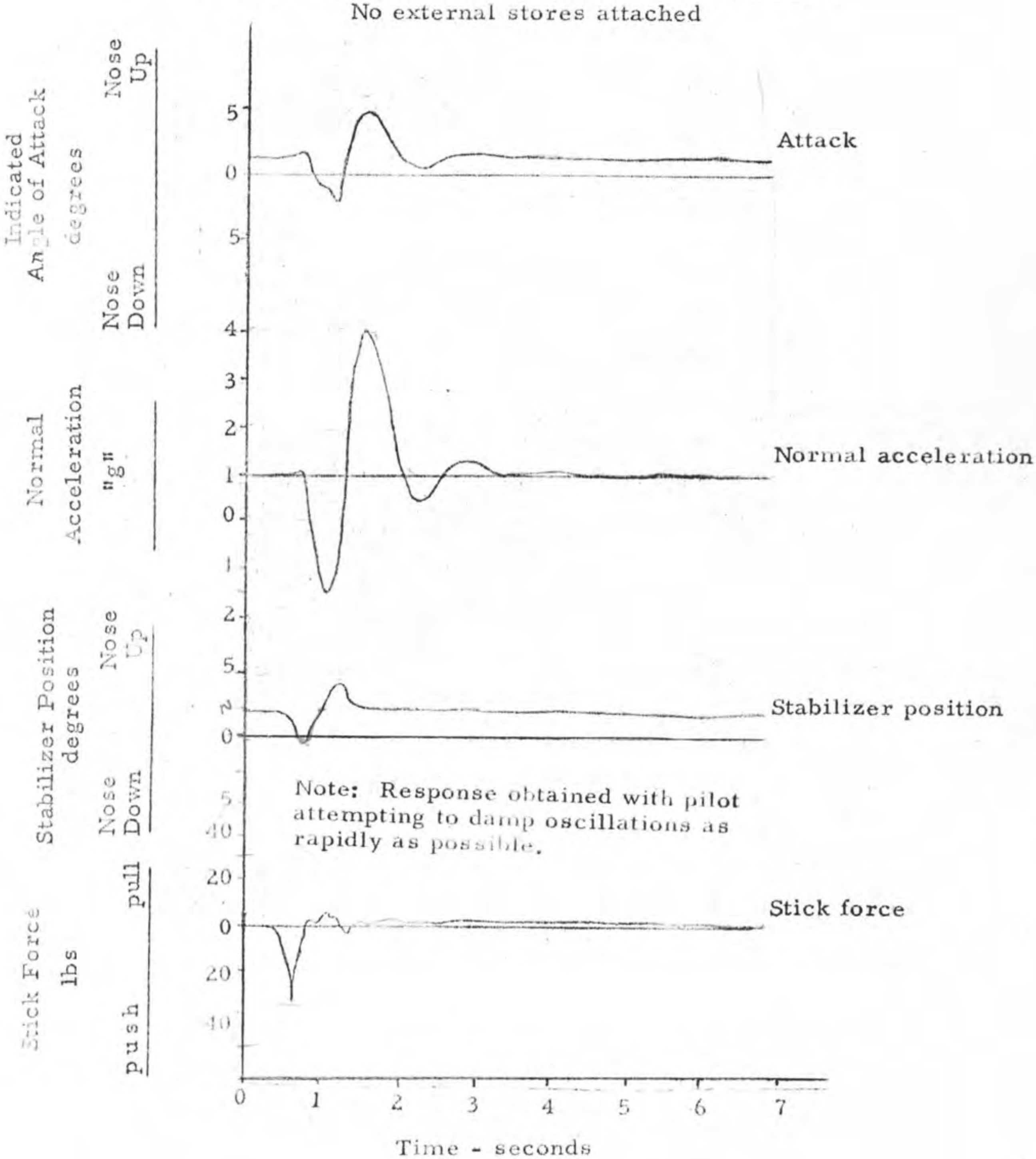
Power Configuration (A/B off)

YF-100A, USAF No. 52-5754

TRIM CONDITIONS

CAS 475 knots	Altitude 10,500 feet
CG 31.2 °/o MAC	Weight 23,600 lbs
Ave N_2 9550 RPM	Stabilizer 1.7° N.U.

No external stores attached



APPENDIX I

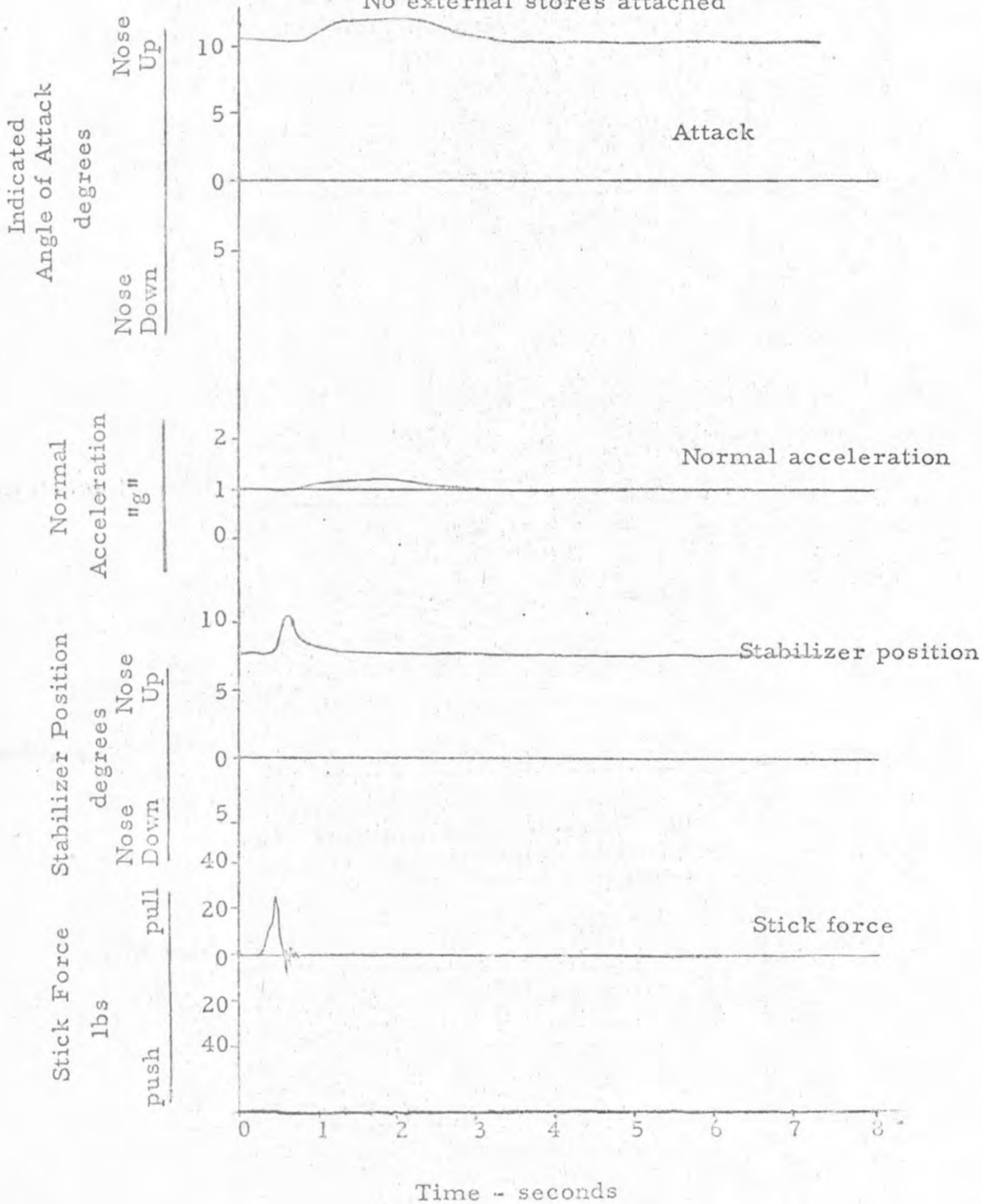
Figure No. 46

DYNAMIC LONGITUDINAL STABILITY
YF-100A, USAF No. 52-5754
Power Approach Configuration Controls Free

TRIM CONDITIONS

CAS 164 knots — Altitude 9,900 feet
CG 30.1 o/o MAC — Weight 21,450 lbs
Ave N_2 9240 RPM — Stabilizer 7.7° N.U.

No external stores attached



APPENDIX I

Figure No. 47

DYNAMIC LONGITUDINAL STABILITY
YF-100A, USAF No. 52-5754
Power Approach Configuration, ~~Controls Free~~

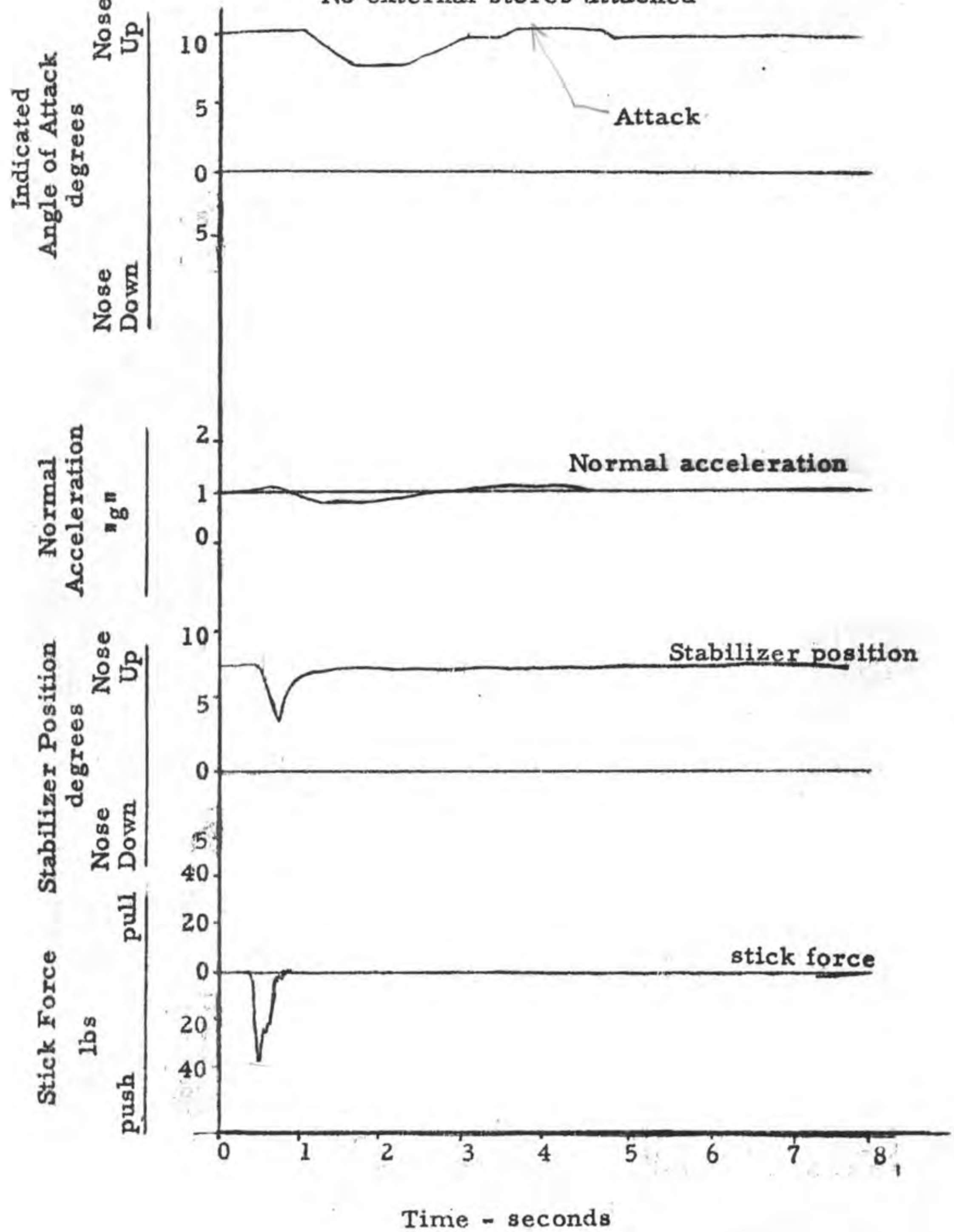
CRUISE
LANDING

TRIM CONDITIONS

IAS 164 knots
CG 30.1 o/o MAC
Ave N_2 9240 RPM

Altitude 9,900 feet
Weight 21,450 lbs
Stabilizer 7.7° N.U.

No external stores attached



APPENDIX I

FIG NO 48

MANUEVERING FLIGHT CHARACTERISTICS

YF-100A USAF No 52-5754

POWER CONFIGURATION (A/B ON)

NO EXTERNAL STORES ATTACHED

TRIM CONDITIONS

CAS 262 KNOTS; CG RANGE 30.5 % MAC

ALT 45800 FEET; AVG WEIGHT 23400 LBS

N_2 3100 RPM; STAB POS 5.6 ° NOSE UP

NOTE: ABRUPT PULL-UP DATA INDICATE 3.2 g's AT 27 POUNDS STICK FORCE FOR 13° NOSE UP STABILIZER

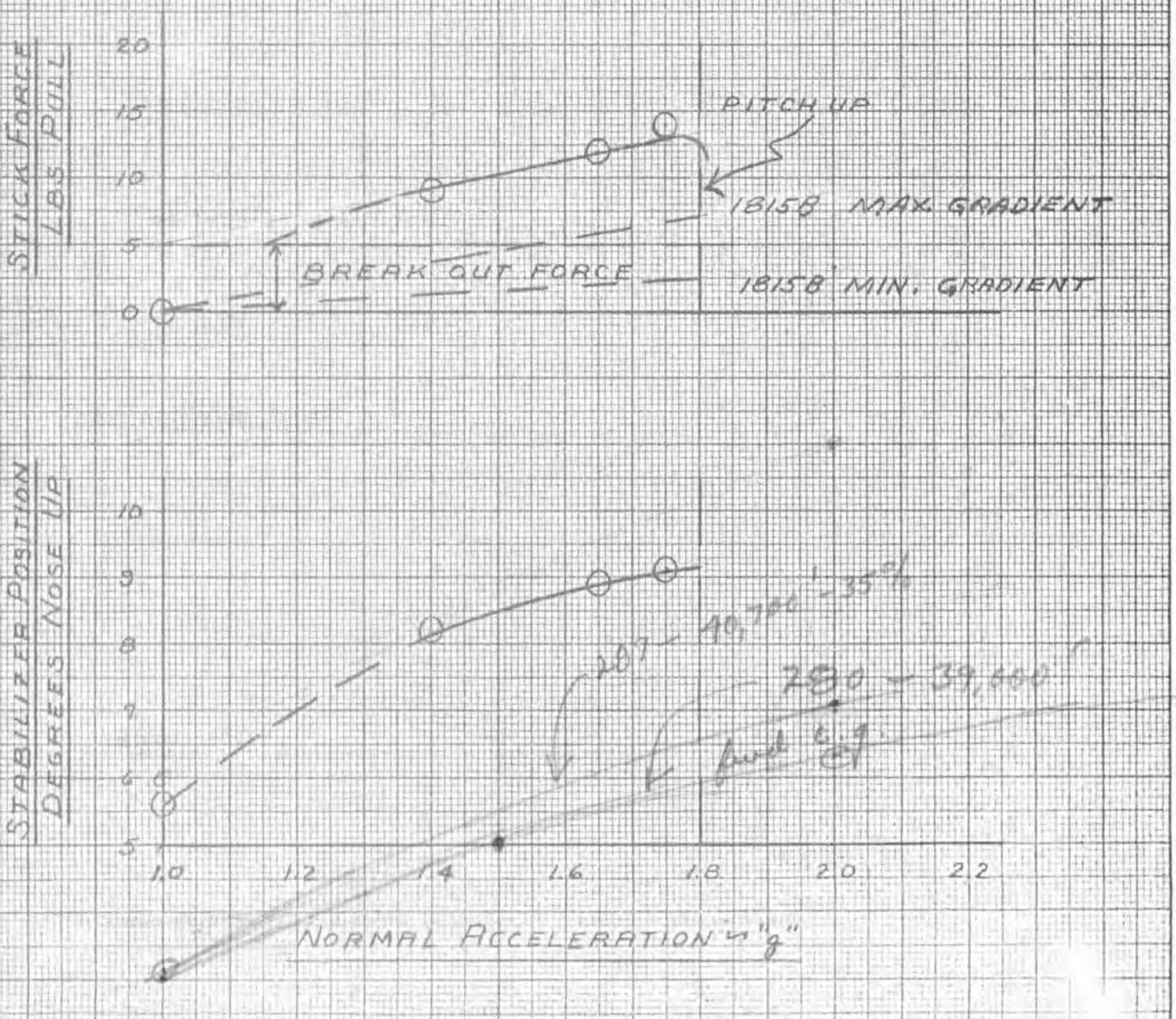


FIG NO 49
MANUEVERING FLIGHT CHARACTERISTICS
YF-100A USAF No 52-5754
POWER CONFIGURATION (A/B ON)
NO EXTERNAL STORES ATTACHED

TRIM CONDITIONS

CAS	271	KNOTS	CG RANGE	29.9	% MAC
ALT	47000	FEET	AVG WEIGHT	22000	LBS
N_2	9055	RPM	STAB POS	5.3	° NOSE UP

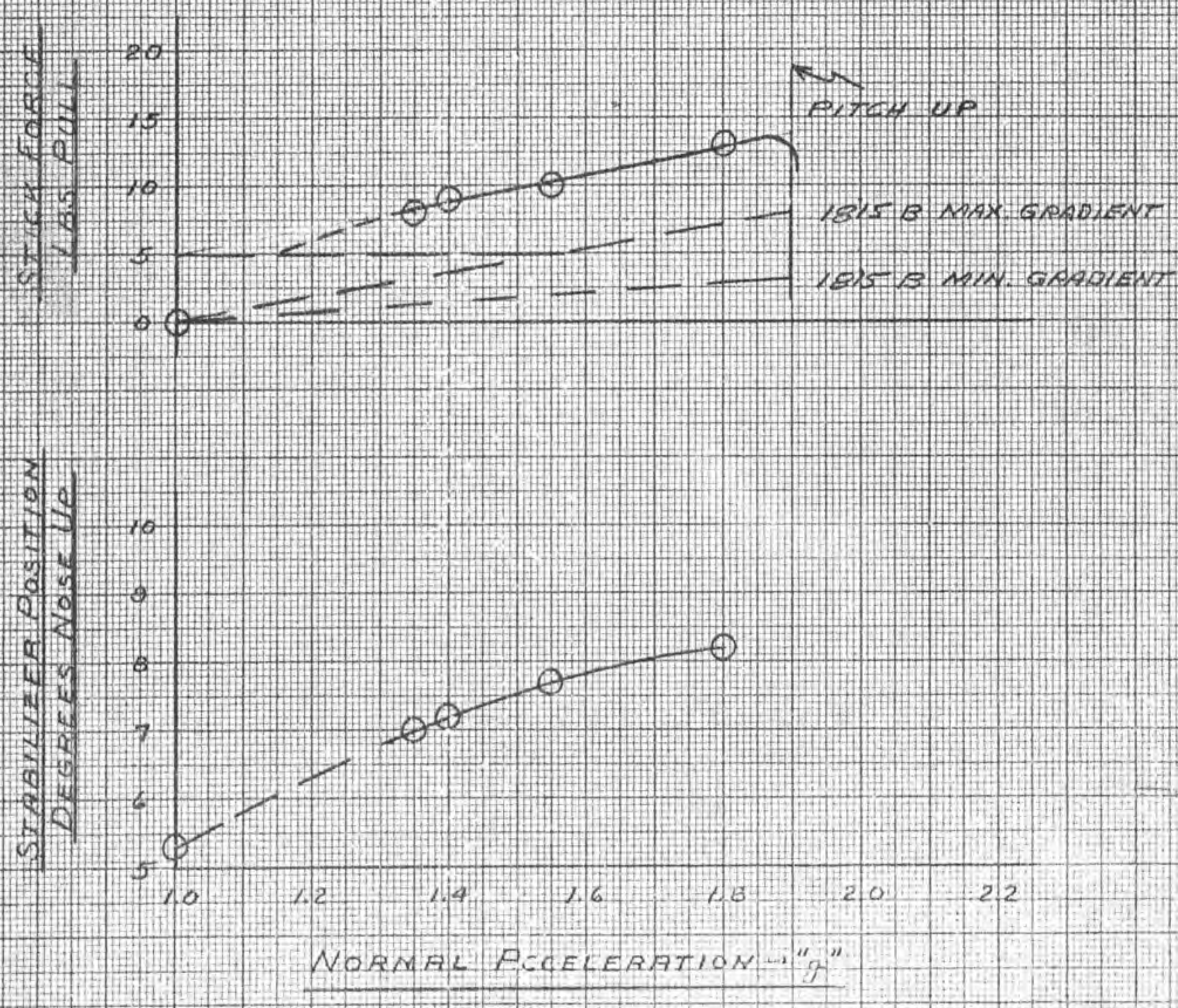


FIG NO 50

MANUEVERING FLIGHT CHARACTERISTICS
YF-100A USAF No 52-5754
CRUISE CONFIGURATION (A/B OFF)
NO EXTERNAL STORES ATTACHED

TRIM CONDITIONS

CAS	254	KNOTS	CG RANGE	29.8	% MAC
ALT	45900	FEET	AVG WEIGHT	21620	LBS
N_2	9000	RPM	STAB POS	4.4	° NOSE UP

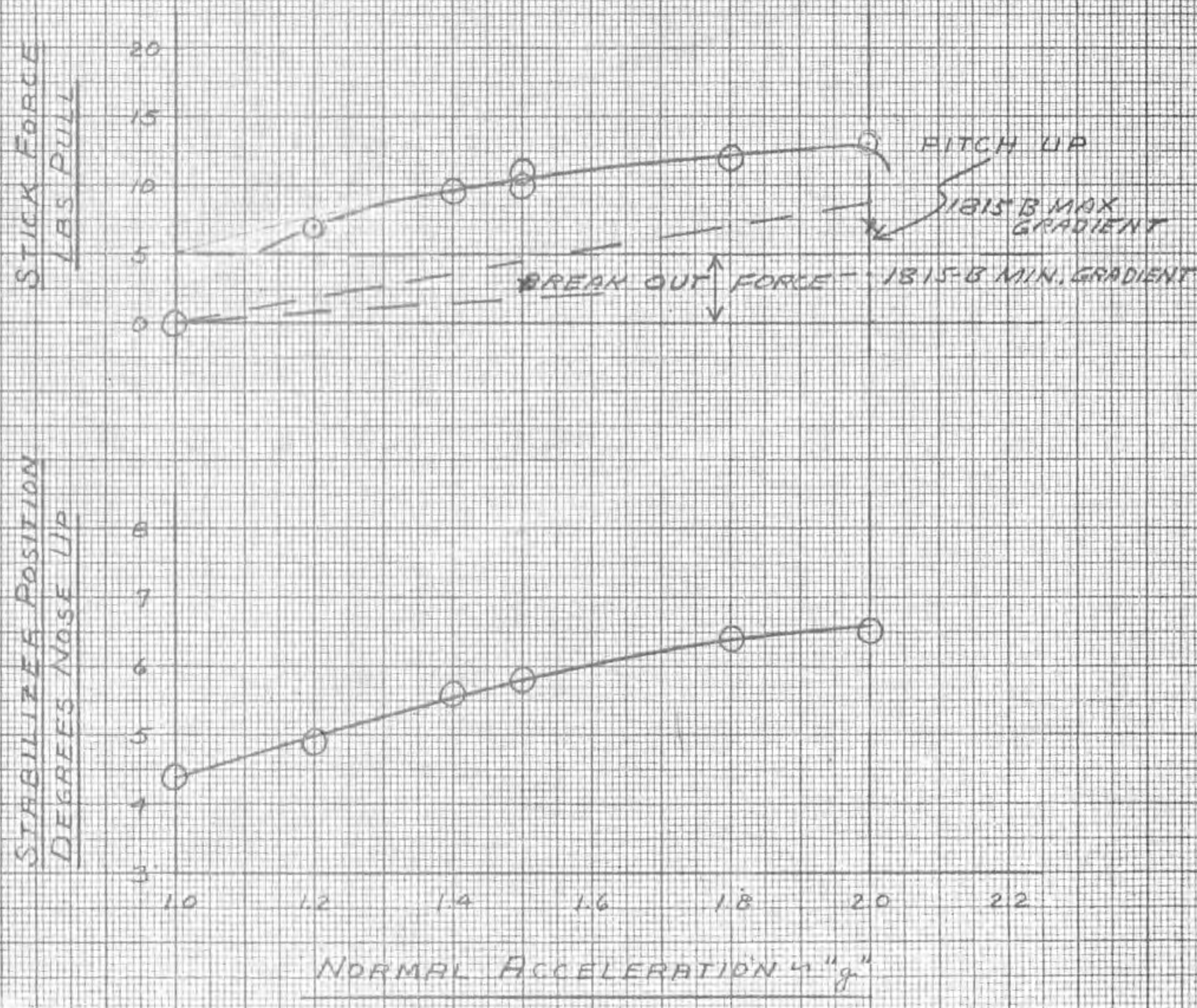


FIG NO 51

MANUEVERING FLIGHT CHARACTERISTICS

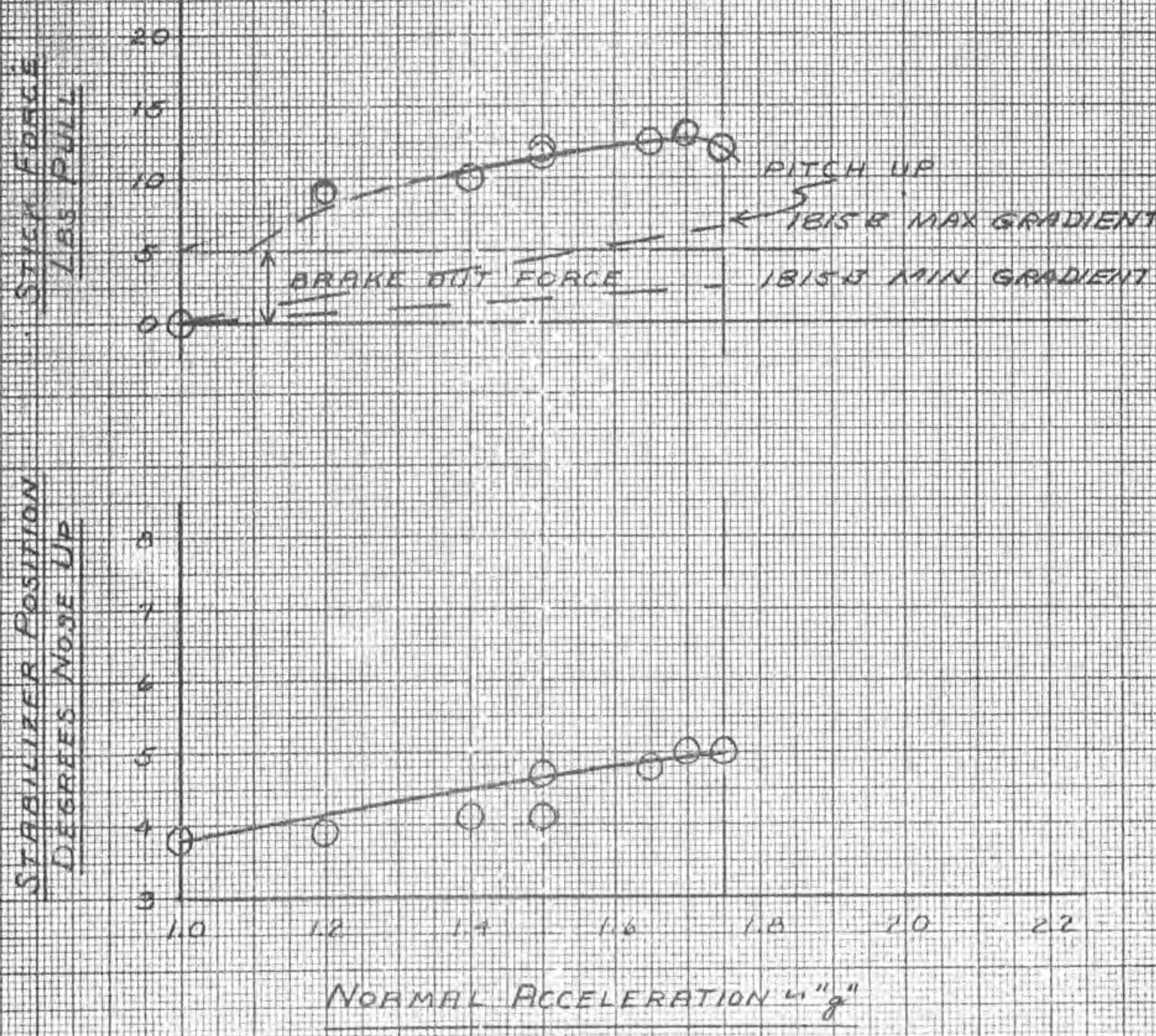
YF-100A USAF No 52-5754

CRUISE CONFIGURATION (A/B OFF)

NO EXTERNAL STORES ATTACHED

TRIM CONDITIONS

CAS	237 KNOTS	CG RANGE	30.35	% MAC
ALT	45700 FEET	AVG WEIGHT	21300	LBS
N_2	8970 RPM	STAB POS	3.8	° NOSE UP



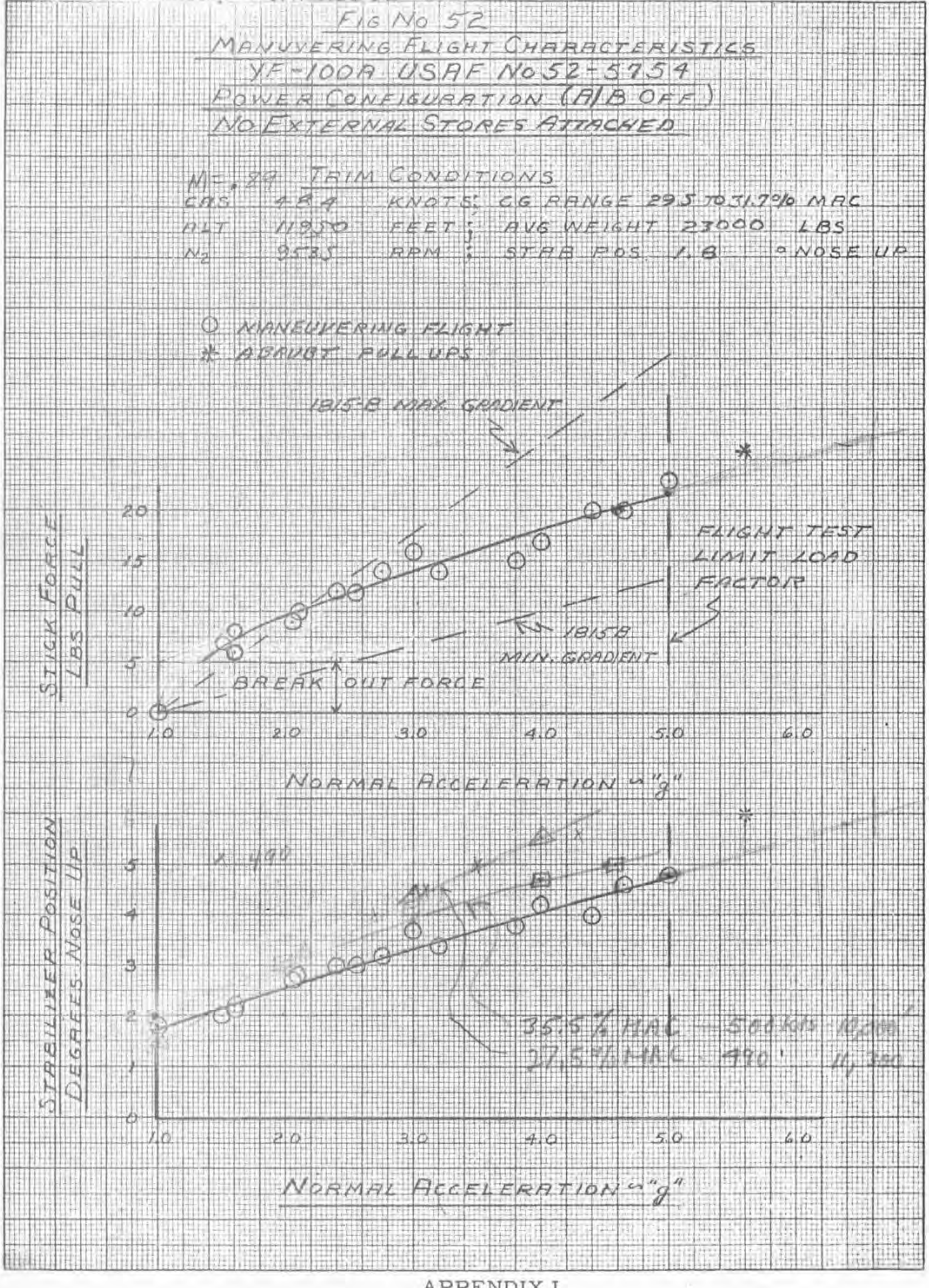


FIG NO 52
MANUVERING FLIGHT CHARACTERISTICS
YF-100A USAF NO 52-5754
POWER CONFIGURATION (A/B OFF)
NO EXTERNAL STORES ATTACHED
M=.79 TRIM CONDITIONS
CAS 484 KNOTS; CG RANGE 29.5 TO 31.7% MAC
ALT 11950 FEET; AVG WEIGHT 23000 LBS
N2 9535 RPM; STAB POS 1.8 ° NOSE UP
○ MANEUVERING FLIGHT
* ABRUPT PULL UPS
1815-B MAX. GRADIENT
FLIGHT TEST LIMIT LOAD FACTOR
1815B MIN. GRADIENT
BREAK OUT FORCE
STICK FORCE LBS PULL
NORMAL ACCELERATION ~ "g"
STABILIZER POSITION DEGREES NOSE UP
35.5% MAC
27.5% MAC
NORMAL ACCELERATION ~ "g"

FIG NO 53

MANUVERING FLIGHT CHARACTERISTICS

YF-100A USAF No 52-5754

POWER APPROACH CONFIGURATION (A/B OFF)

NO EXTERNAL STORES ATTACHED

TRIM CONDITIONS

IAS	159	KNOTS	CG RANGE	31	% MAC
ALT	11190	FEET	AVG WEIGHT	21900	LBS
N_2	9220	RPM	STAB POS	7.9	°NOSE UP

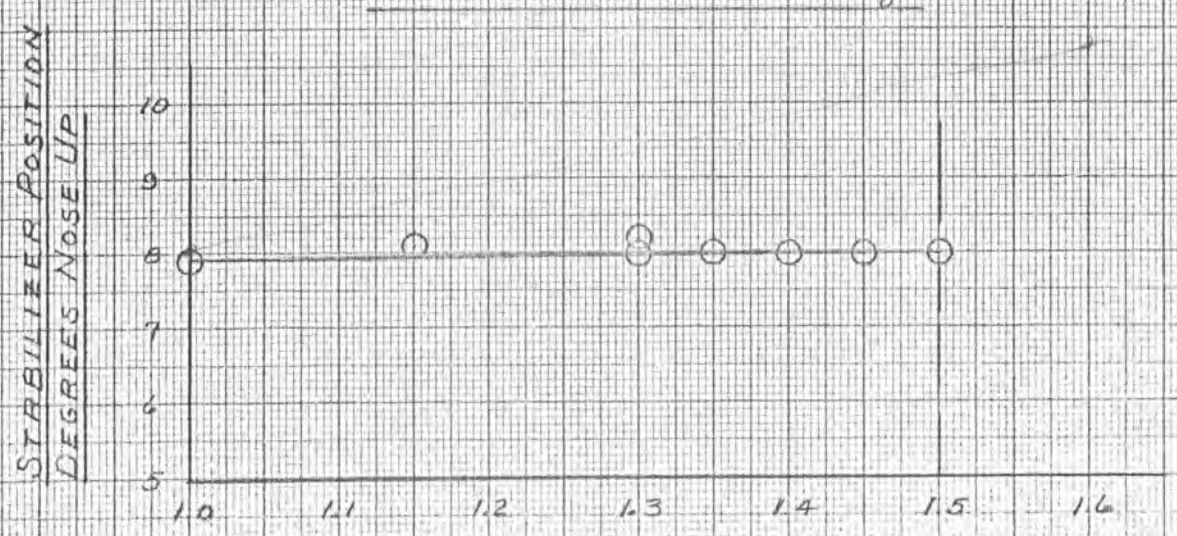
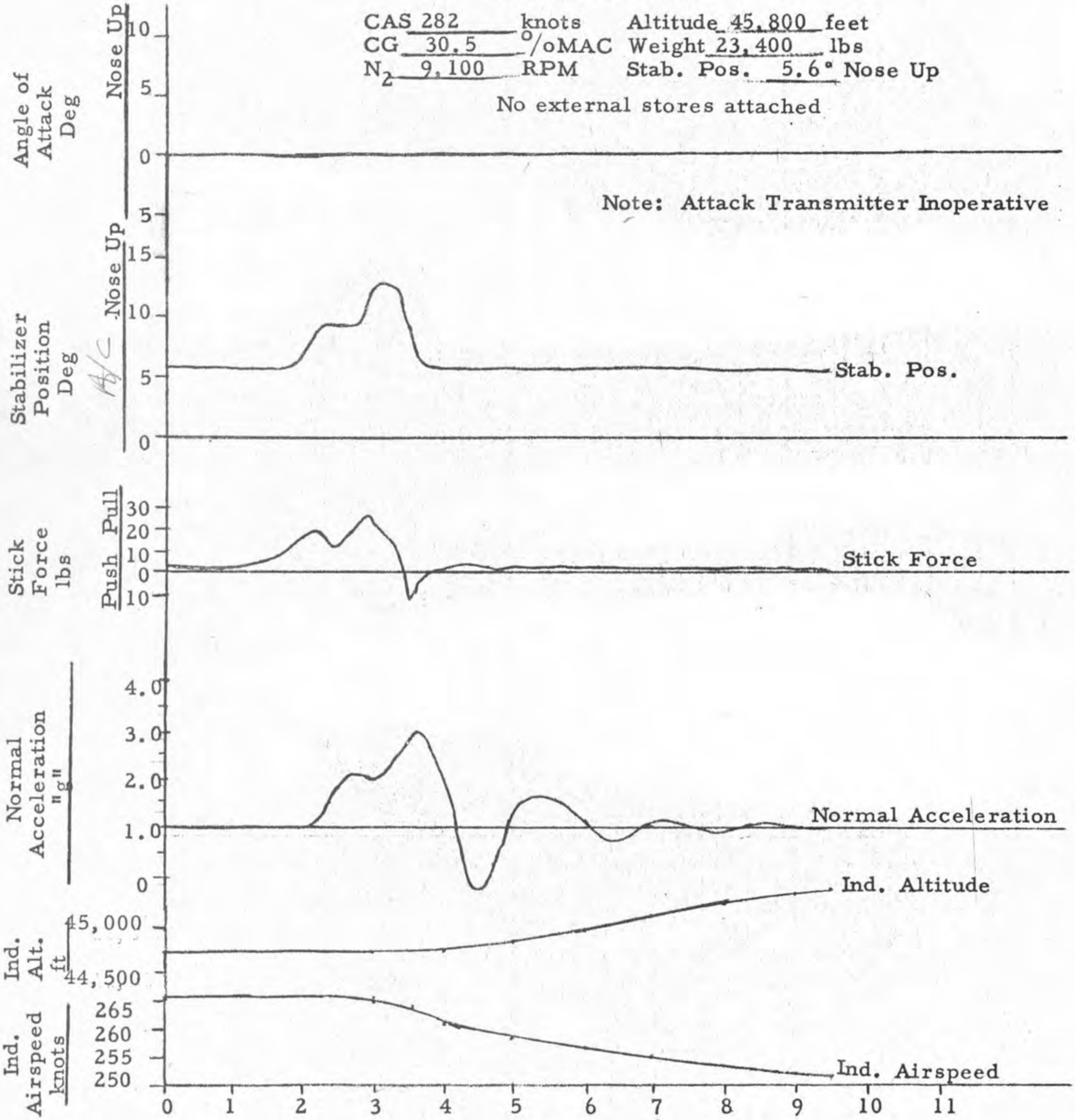


Figure No. 54
TIME HISTORY OF AN ABRUPT PULL-UP
YF-100A, USAF NO. 52-5754
Power Configuration (A/B On)
TRIM CONDITIONS



Time - Seconds

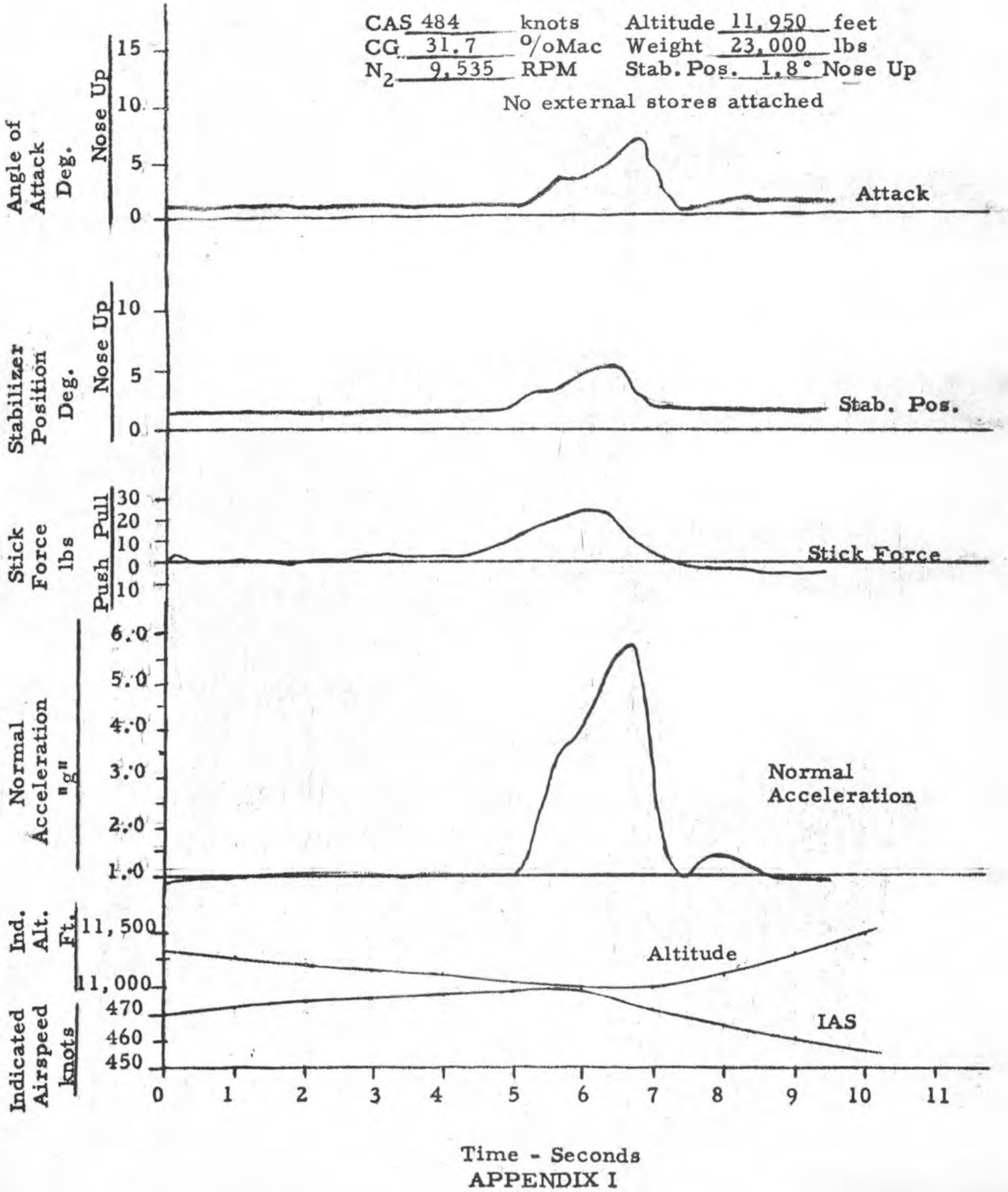
Figure No. 55

TIME HISTORY OF AN ABRUPT PULL-UP
YF-100A, USAF No. 52-5754
Power Configuration (A/B Off)

TRIM CONDITIONS

CAS 484	knots	Altitude 11,950	feet
CG 31.7	°/oMac	Weight 23,000	lbs
N_2 9,535	RPM	Stab. Pos. 1.8°	Nose Up

No external stores attached

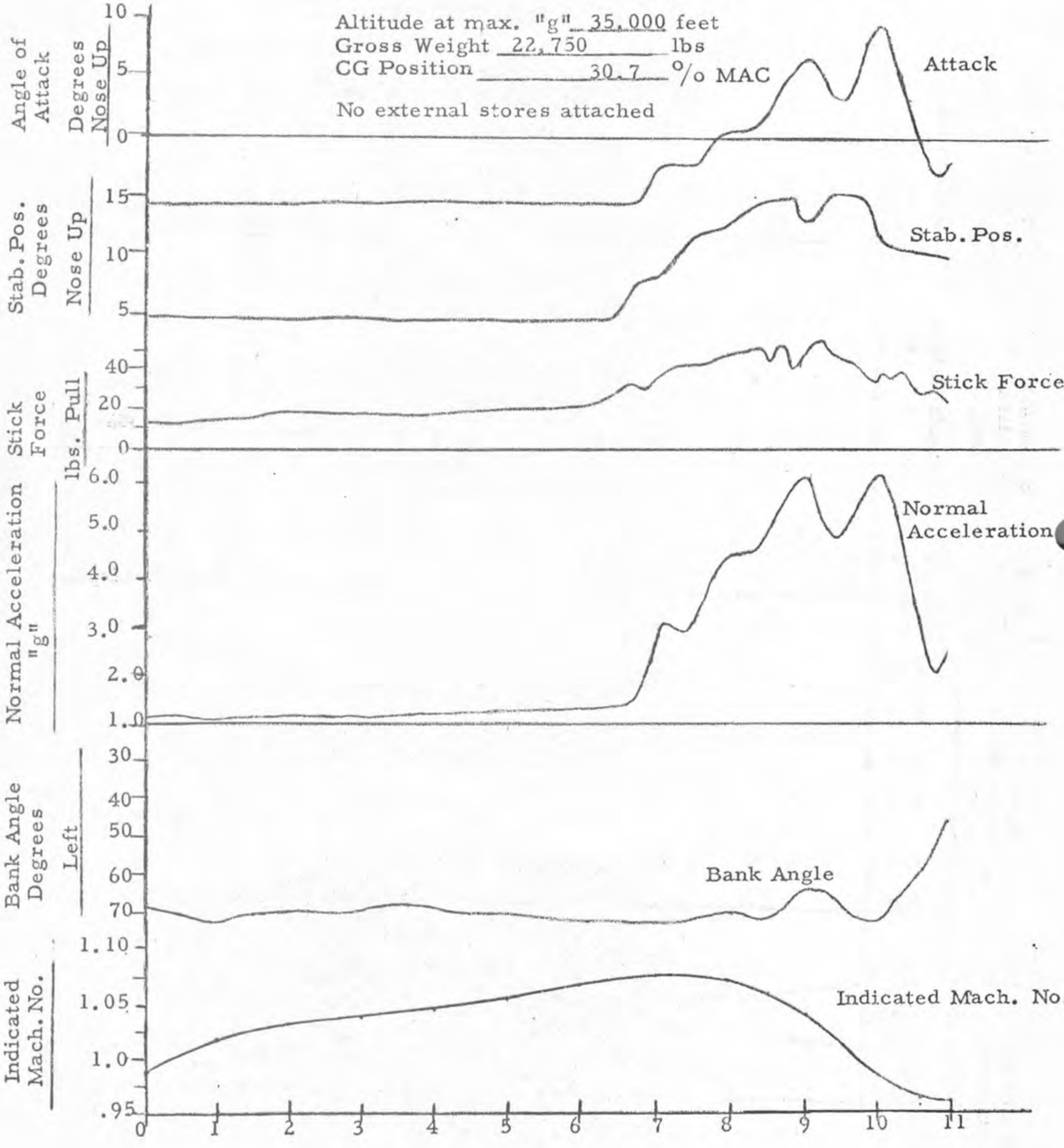


Time - Seconds

APPENDIX I

Figure No. 56

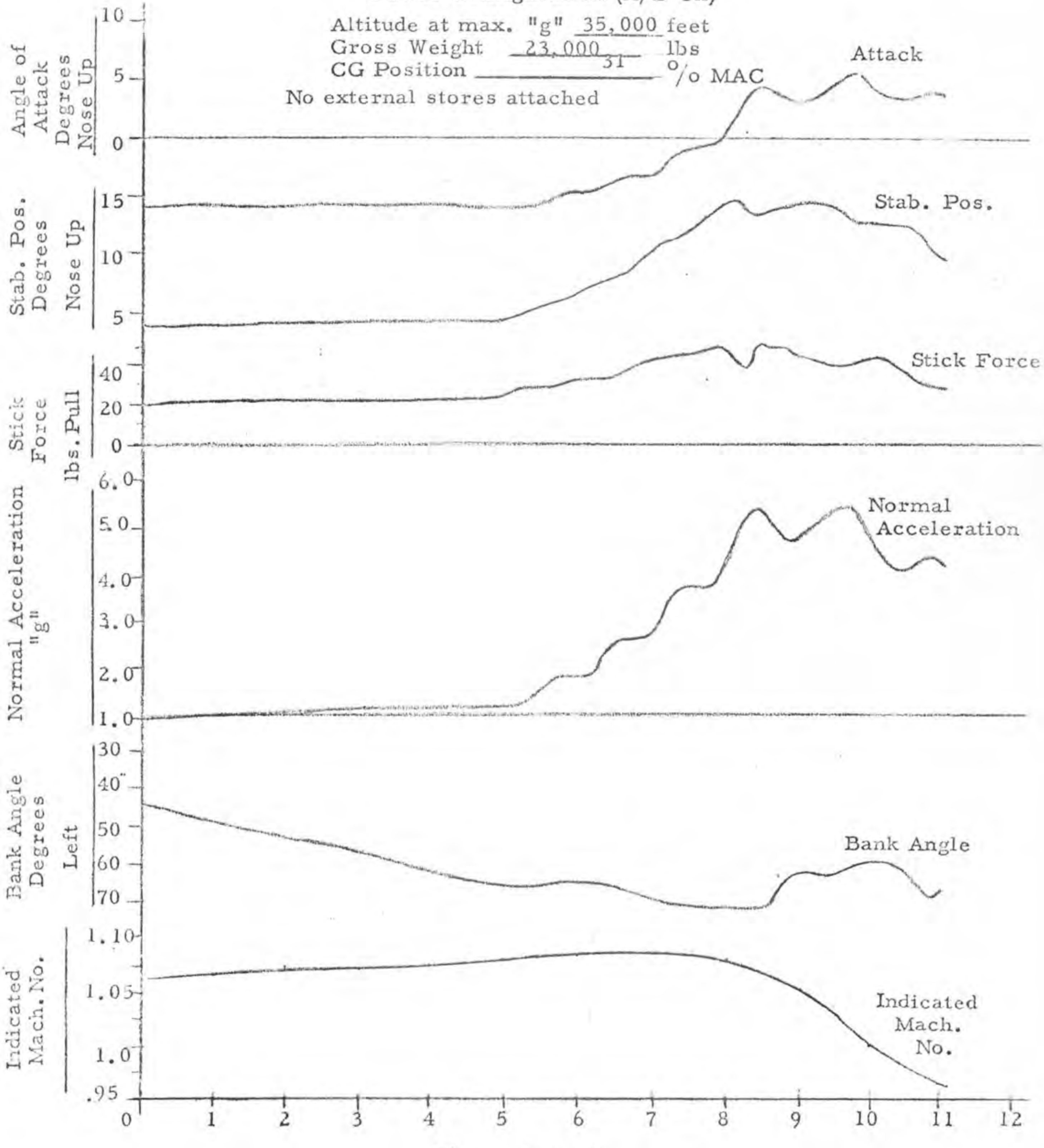
TIME HISTORY OF LIMIT "g" DIVING TURN
YF-100A, USAF No. 52-5754
Power Configuration (A/B On)



APPENDIX I

Figure No. 57

TIME HISTORY OF LIMIT "g" DIVING TURN
YF-100A, USAF No. 52-5754
Power Configuration (A/B On)



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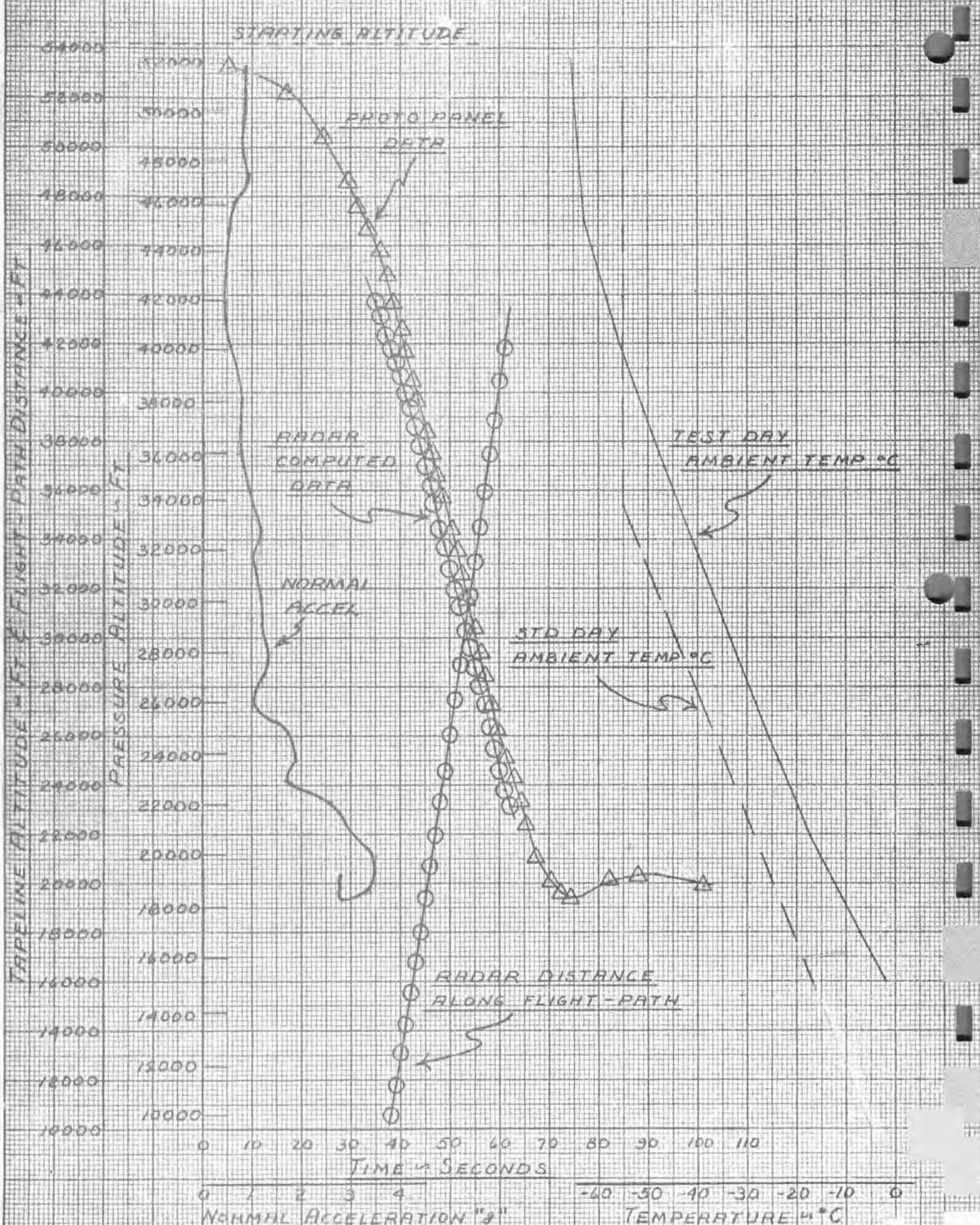
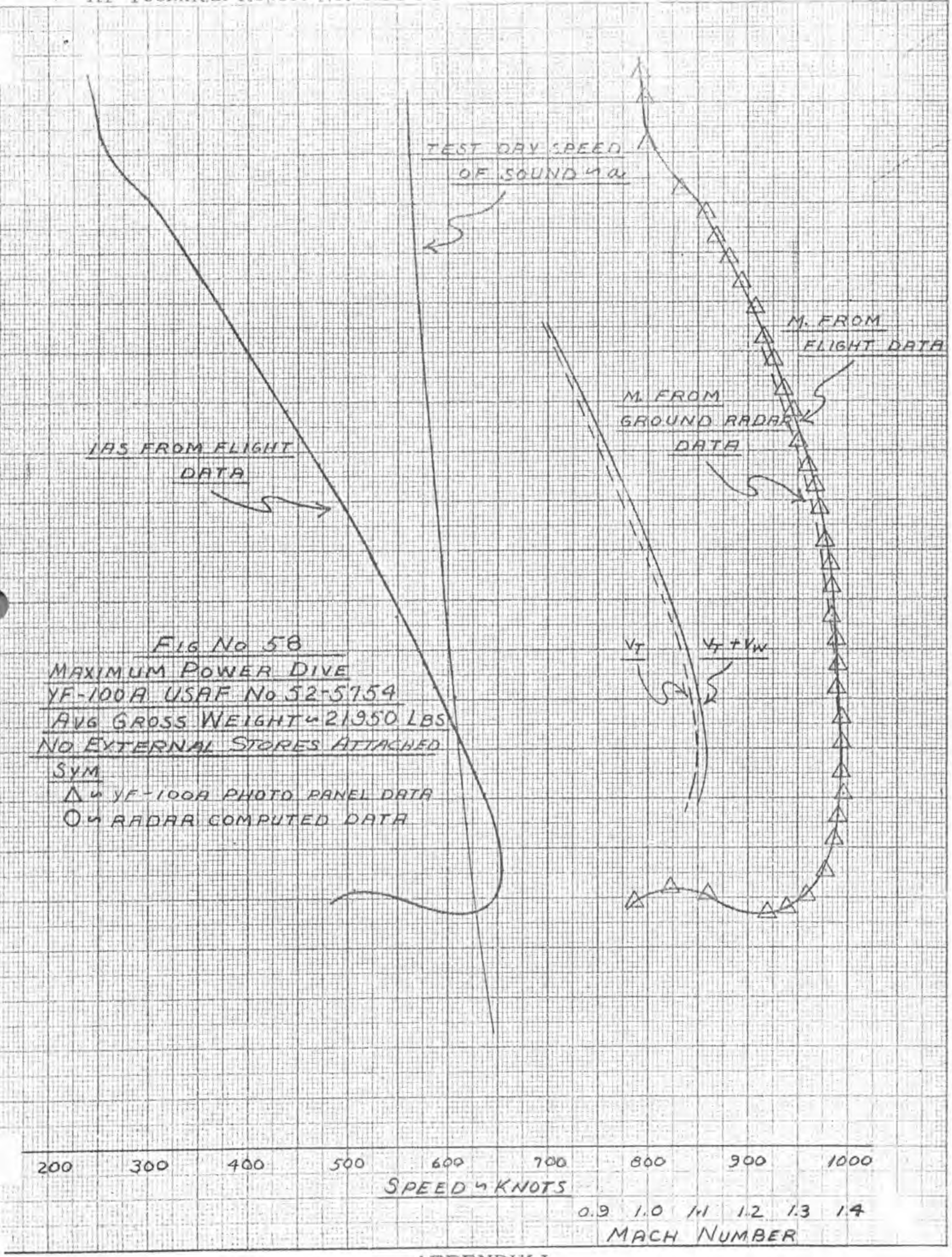

STARTING ALTITUDE
PHOTO PANEL DATA
RADAR COMPUTED DATA
NORMAL ACCEL
STD DAY AMBIENT TEMP °C
TEST DAY AMBIENT TEMP °C
RADAR DISTANCE ALONG FLIGHT-PATH
TAPELINE ALTITUDE - FT & FLIGHT-PATH DISTANCE - FT
PRESSURE ALTITUDE - FT
TIME - SECONDS
NORMAL ACCELERATION "g"
TEMPERATURE - °C


TEST DAY SPEED OF SOUND ~ a
IAS FROM FLIGHT DATA
M. FROM FLIGHT DATA
M. FROM GROUND RADAR DATA
V_T
$V_T + V_W$
FIG NO 58
MAXIMUM POWER DIVE
YF-100A USAF No 52-5754
AVG GROSS WEIGHT ~ 21950 LBS
NO EXTERNAL STORES ATTACHED
SYM
△ ~ YF-100A PHOTO PANEL DATA
○ ~ RADAR COMPUTED DATA
200 300 400 500 600 700 800 900 1000
SPEED ~ KNOTS
0.9 1.0 1.1 1.2 1.3 1.4
MACH NUMBER

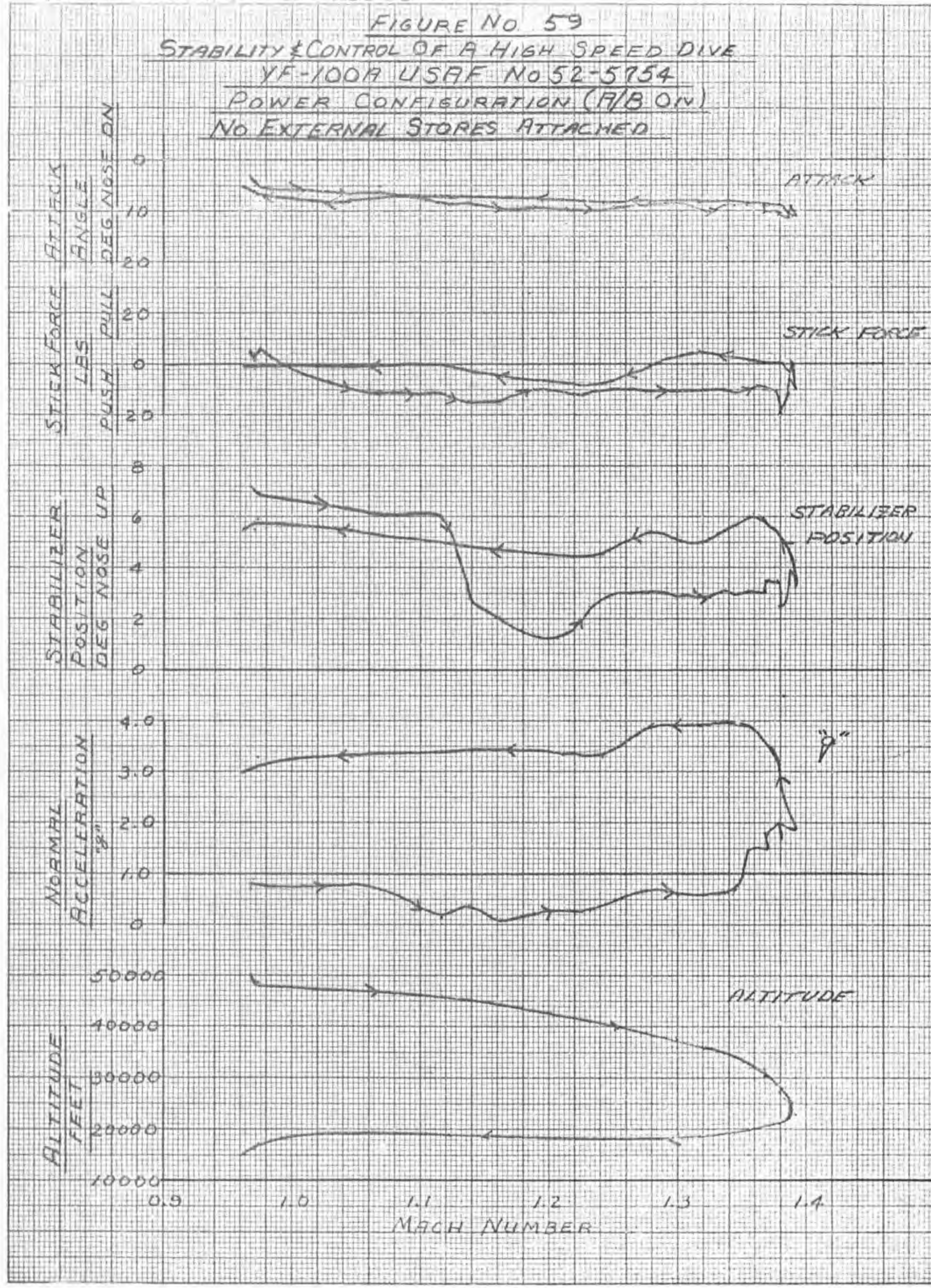
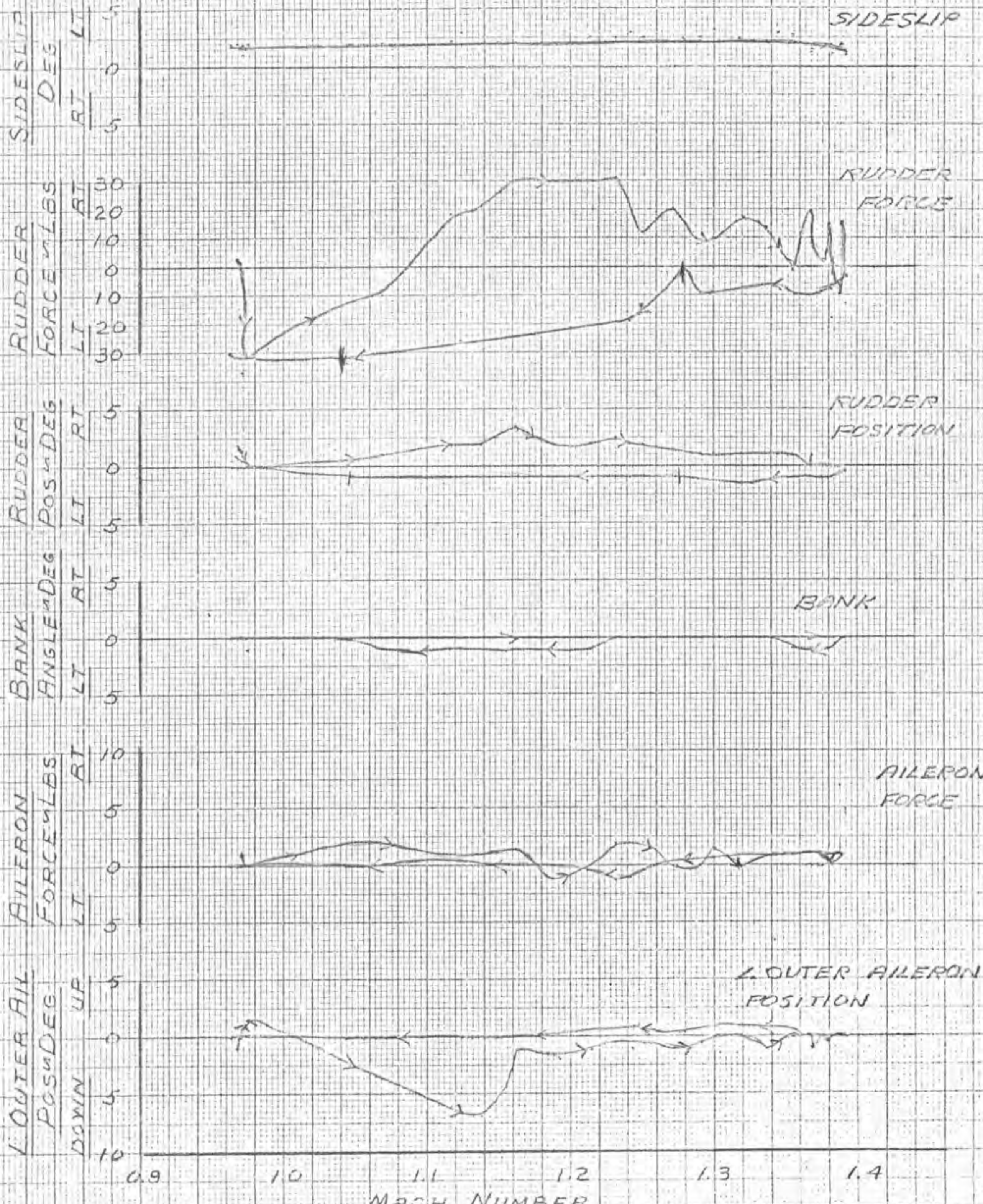


FIGURE NO. 59
STABILITY & CONTROL OF A HIGH SPEED DIVE
YF-100A USAF No 52-5754
POWER CONFIGURATION (A/B ON)
NO EXTERNAL STORES ATTACHED
ATTACK ANGLE DEG NOSE DN
0
10
20
ATTACK
STICK FORCE LBS PUSH PULL
20
0
20
STICK FORCE
STABILIZER POSITION DEG NOSE UP
8
6
4
2
0
STABILIZER POSITION
NORMAL ACCELERATION "g"
4.0
3.0
2.0
1.0
0
"g"
ALTITUDE FEET
50000
40000
30000
20000
10000
ALTITUDE
0.9
1.0
1.1
1.2
1.3
1.4
MACH NUMBER

FIGURE NO 59

TRIM CONDITIONS

IAS 222 KNOTS CG 29.62 MAC RUDDER POS 1.3° RIGHT
ALTITUDE 51200 FT N_2 8070 RPM L OUTER AIL. POS 0.8° DN
WEIGHT 21700 LBS STAB POS 6.8° NOSE UP



AF Technical Report No.
AFFTC 53-33

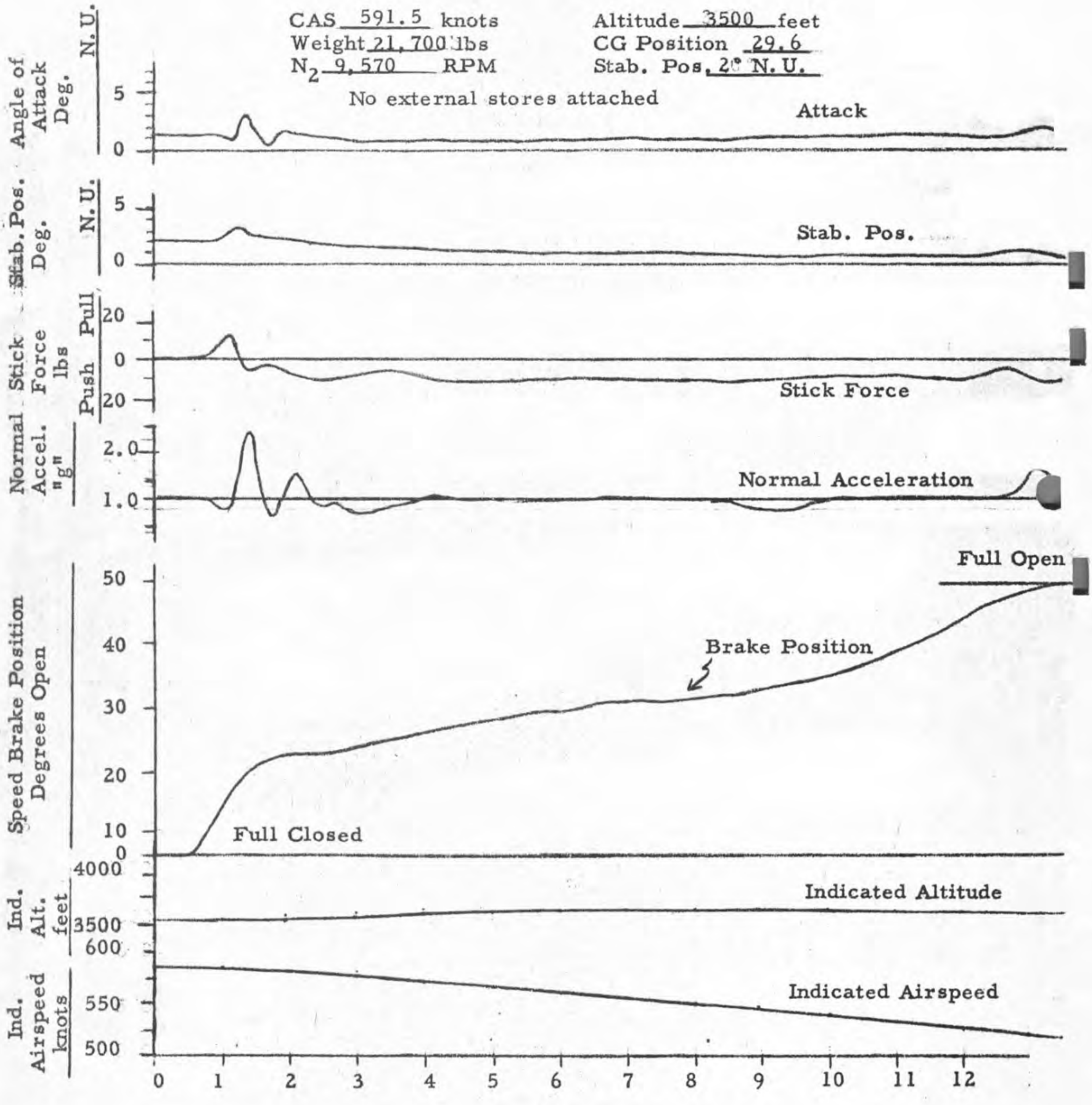
Figure No. 60

TIME HISTORY OF SPEED BRAKE OPENING

YF-100A, USAF No. 52-5754

Power Configuration

TRIM CONDITIONS



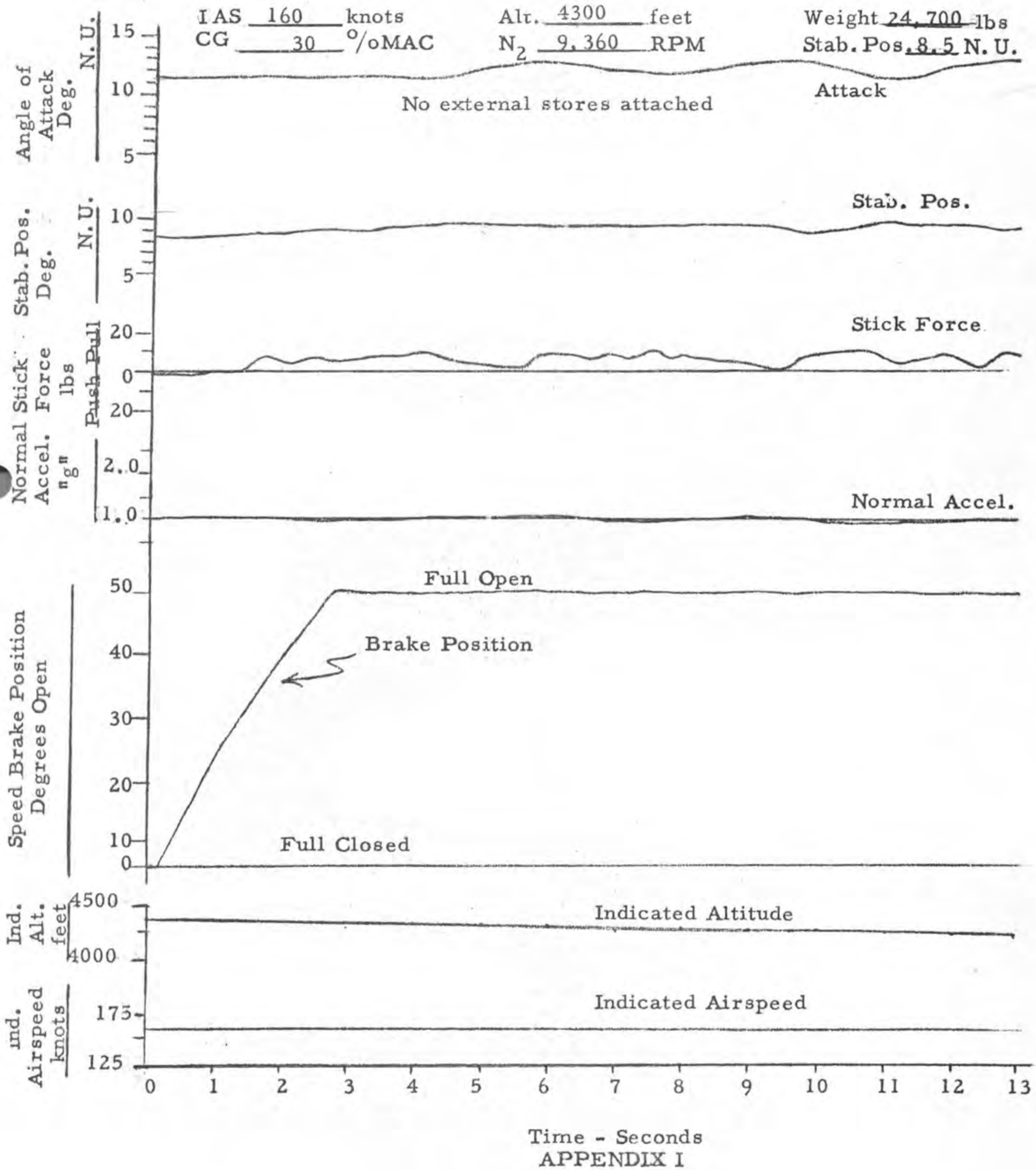
APPENDIX I

Figure No. 61

TIME HISTORY OF SPEED BRAKE OPENING

YF-100 A, USAF No. 52-5754
Power Approach Configuration

TRIM CONDITIONS



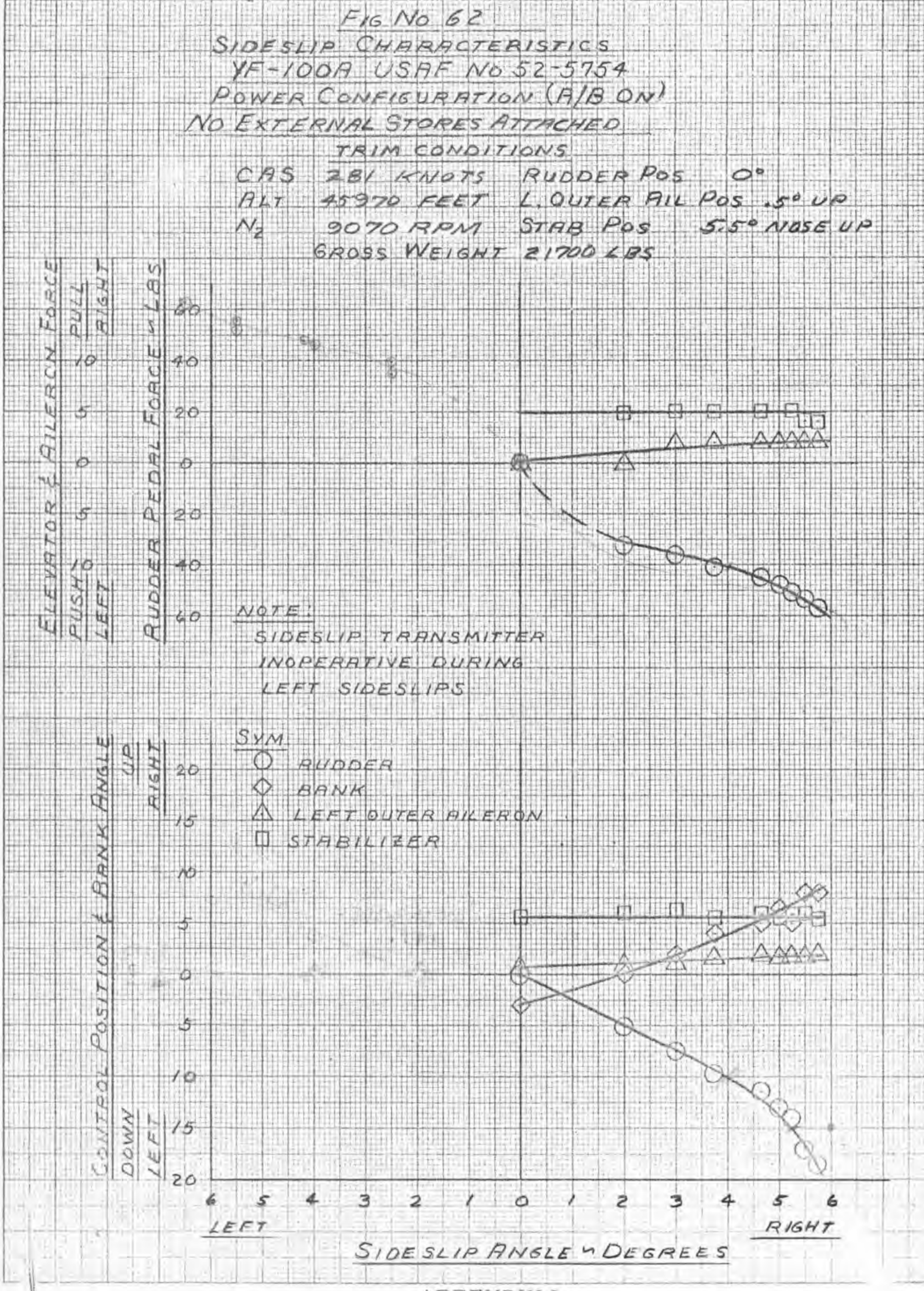


FIG NO 62
SIDESLIP CHARACTERISTICS
YF-100A USAF NO 52-5754
POWER CONFIGURATION (A/B ON)
NO EXTERNAL STORES ATTACHED
TRIM CONDITIONS
CAS 281 KNOTS RUDDER POS 0°
ALT 45970 FEET L. OUTER AIL POS .5° UP
N2 9070 RPM STAB POS 5.5° NOSE UP
GROSS WEIGHT 21700 LBS
ELEVATOR & AILERON FORCE
PUSH LEFT
PULL RIGHT
RUDDER PEDAL FORCE ~ LBS
NOTE:
SIDESLIP TRANSMITTER INOPERATIVE DURING LEFT SIDESLIPS
SYM
RUDDER
BANK
LEFT OUTER AILERON
STABILIZER
CONTROL POSITION & BANK ANGLE
DOWN LEFT
UP RIGHT
LEFT
RIGHT
SIDESLIP ANGLE ~ DEGREES

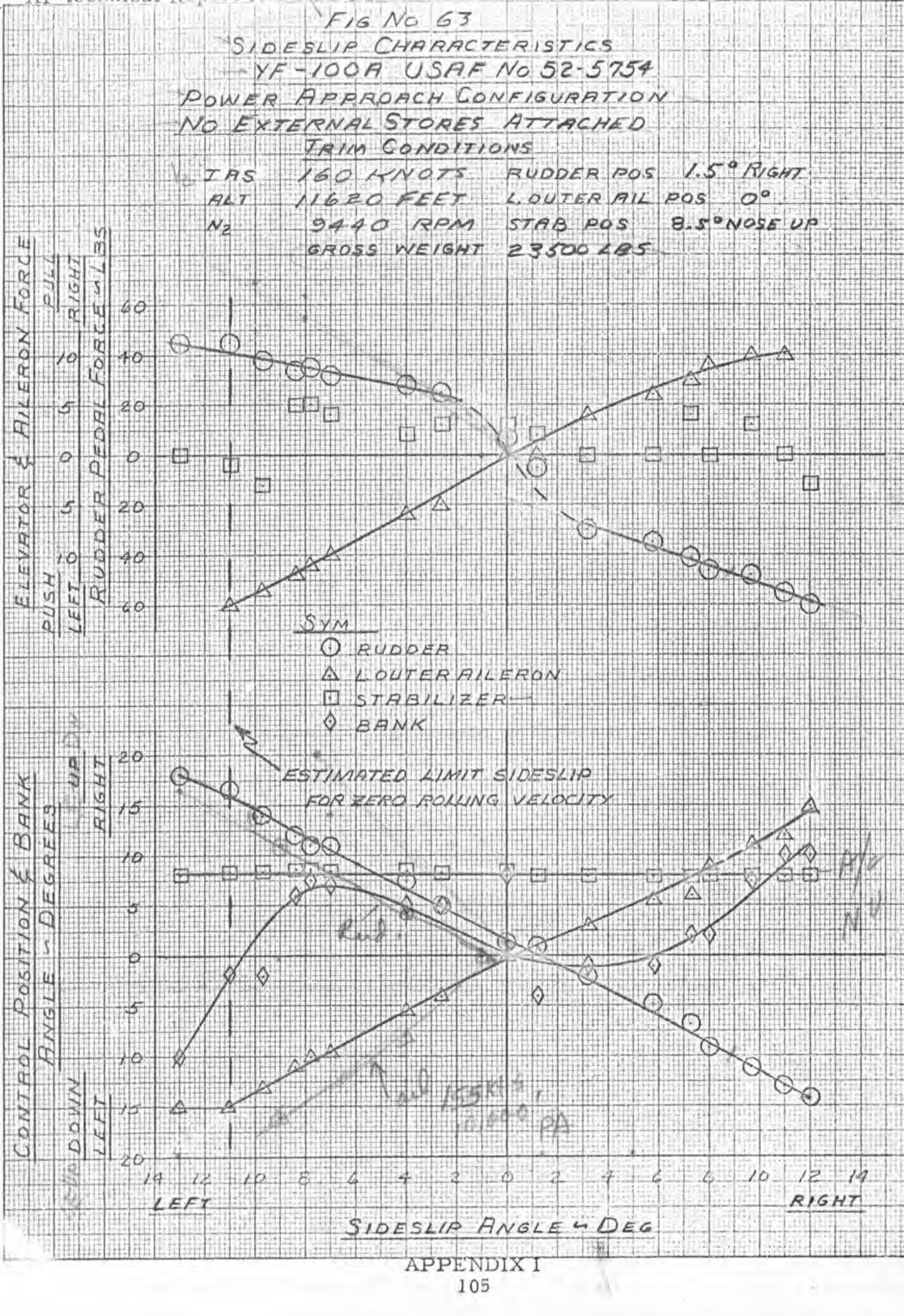


FIG NO 63
SIDESLIP CHARACTERISTICS
YF-100A USAF No 52-5754
POWER APPROACH CONFIGURATION
NO EXTERNAL STORES ATTACHED
TRIM CONDITIONS
IAS 160 KNOTS
ALT 11620 FEET
N2 9440 RPM
GROSS WEIGHT 23500 LBS
RUDDER POS 1.5° RIGHT
L. OUTER AIL POS 0°
STAB POS 8.5° NOSE UP
ELEVATOR & AILERON FORCE
PUSH PULL
LEFT RIGHT
RUDDER PEDAL FORCE ~ LBS
SYM
RUDDER
L OUTER AILERON
STABILIZER
BANK
ESTIMATED LIMIT SIDESLIP FOR ZERO ROLLING VELOCITY
CONTROL POSITION & BANK ANGLE - DEGREES
DOWN UP
LEFT RIGHT
LEFT
RIGHT
SIDESLIP ANGLE ~ DEG

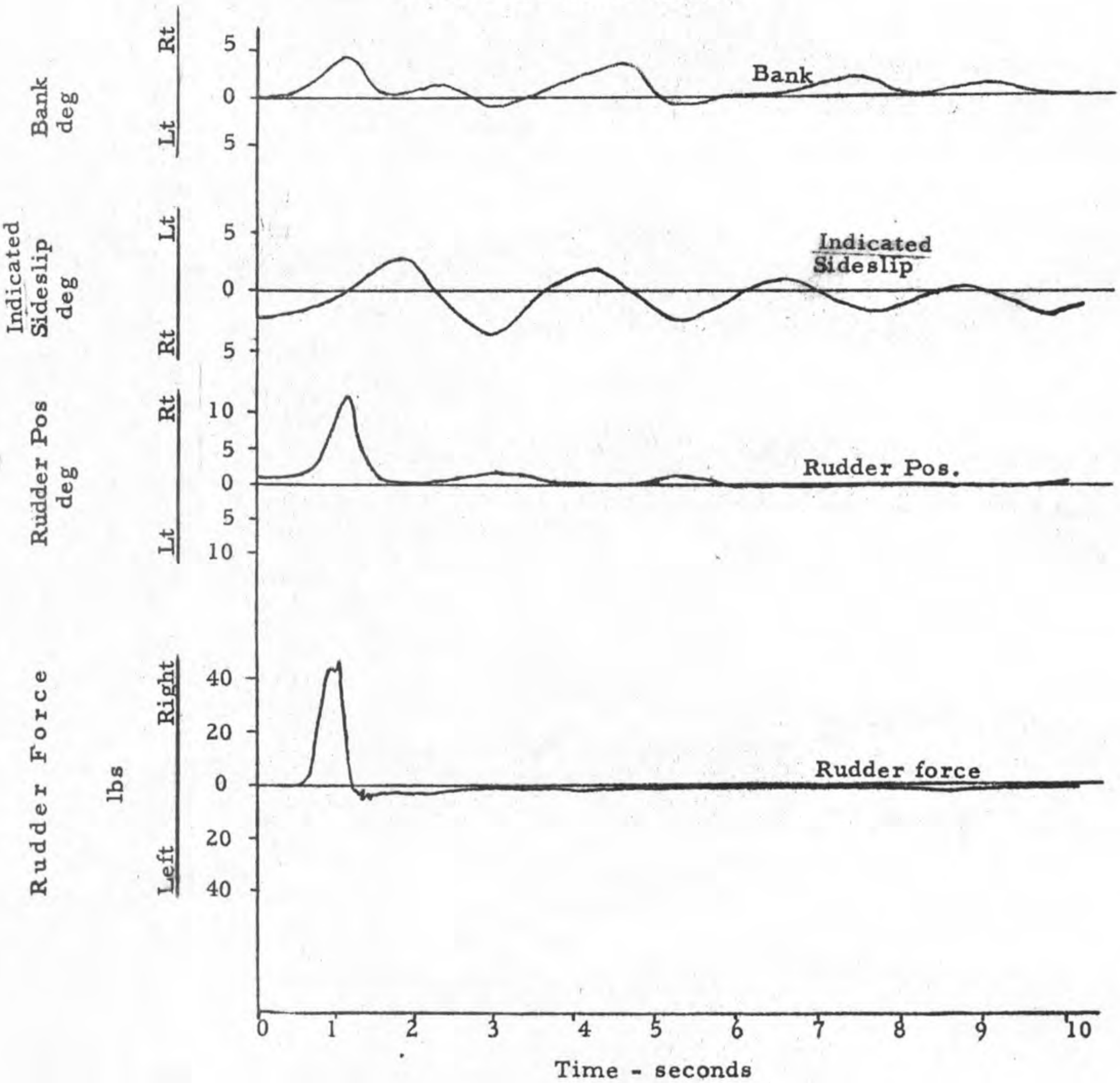
Figure No. 64

DYNAMIC DIRECTIONAL STABILITY
YF-100A, USAF No. 52-5754
Power Configuration (A/B On) Controls Free

TRIM CONDITIONS

CAS 274 knots Altitude 45,300 feet
CG 29.8 o/o MAC Weight 22,200 lbs
Ave N_2 9070 RPM Rudder Pos. 0 degrees
L. Aileron Position 0.4 degrees Up

No external stores attached



APPENDIX I

Figure No. 65

DYNAMIC DIRECTIONAL STABILITY
YF-100A, USAF No. 52-5754
Power Configuration (A/B On) Controls Free

TRIM CONDITIONS

CAS 274 knots — Altitude 45,300 feet
CG 29.8 °/o MAC — Weight 22,200 lbs
Ave N_2 9070 RPM — Rudder Pos. 0 degrees
L. Aileron Position 0.4 degrees Up

No external stores attached

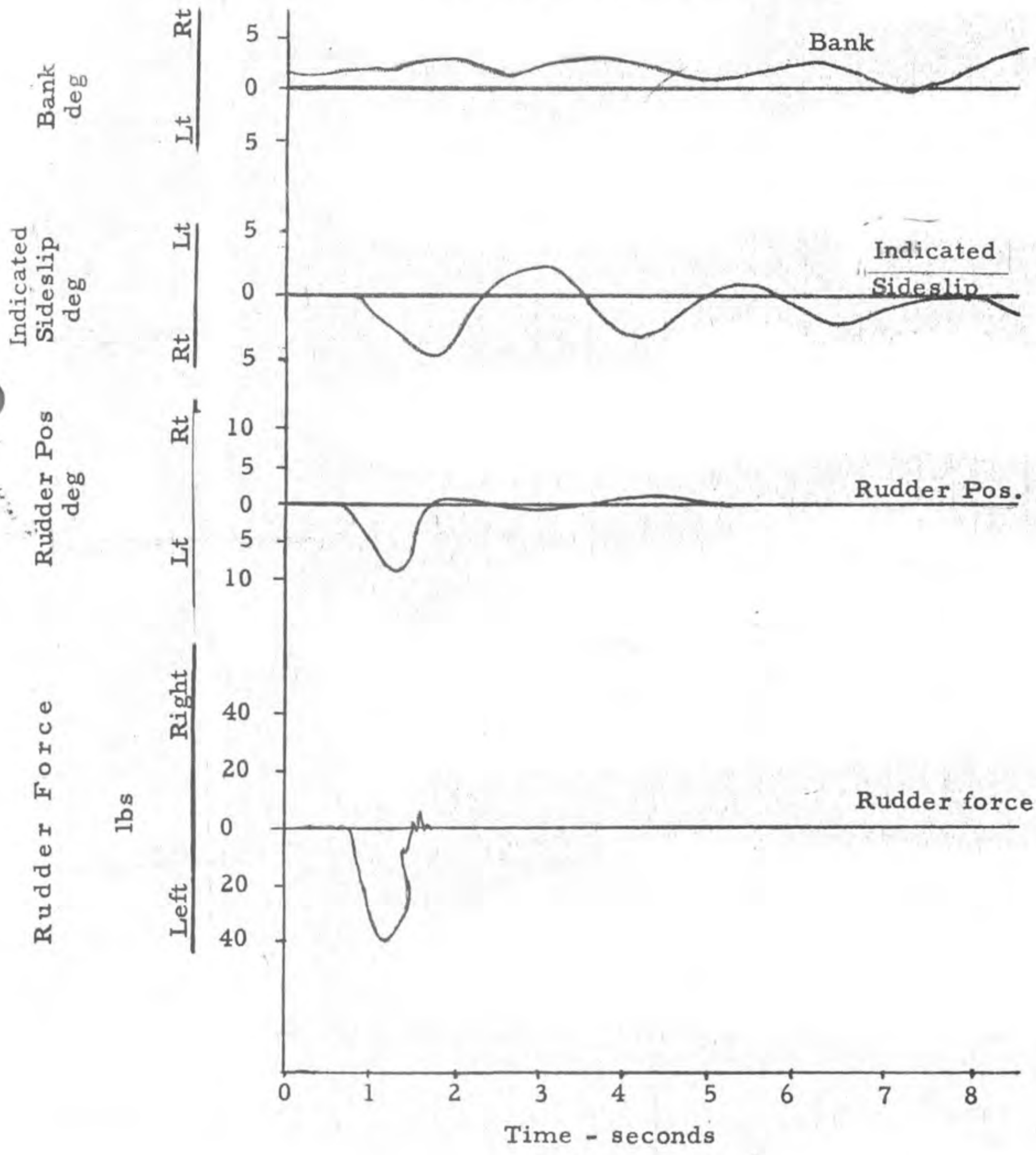


Figure No. 66

DYNAMIC DIRECTIONAL STABILITY
YF-100A, USAF No. 52-5754
Power Configuration(A/B on)

TRIM CONDITIONS

CAS 274 knots Altitude 45,300 feet
CG 29.8 $^o/_o$ MAC Weight 22,200 lbs
Ave N_2 9070 RPM Rudder Pos. 0 degrees
L. Aileron Position 0.4 degrees Up
No external stores attached

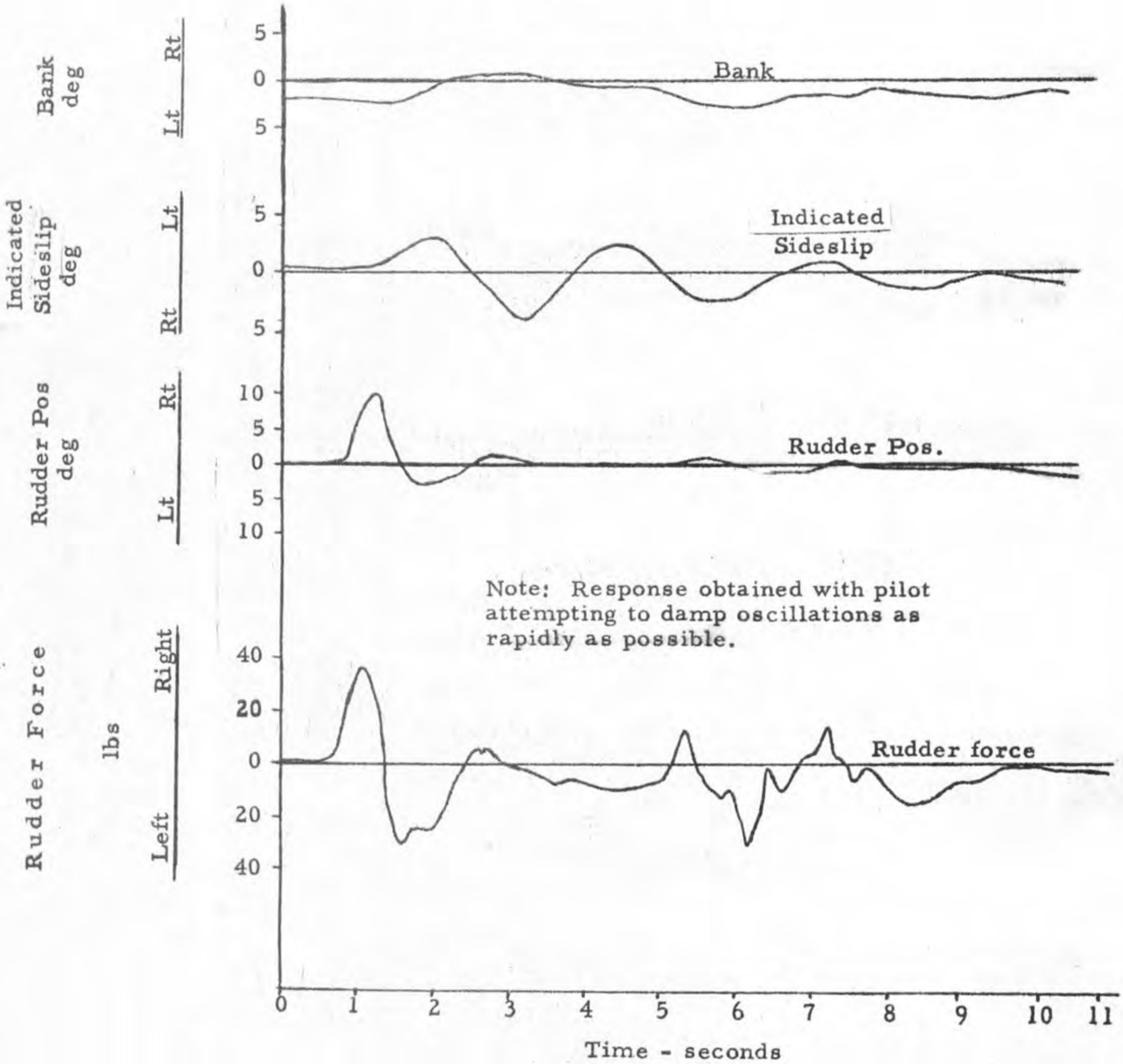


Figure No. 67

DYNAMIC DIRECTIONAL STABILITY
YF-100A, USAF No. 52-5754
Power Configuration (A/B on)

TRIM CONDITIONS

CAS	274	knots	Altitude	45,300	feet
CG	29.8	o/o MAC	Weight	22,200	lbs
Ave N_2	9070	RPM	Rudder Pos.	0	degrees

L. Aileron Position 0.4 degrees Up

No external stores attached

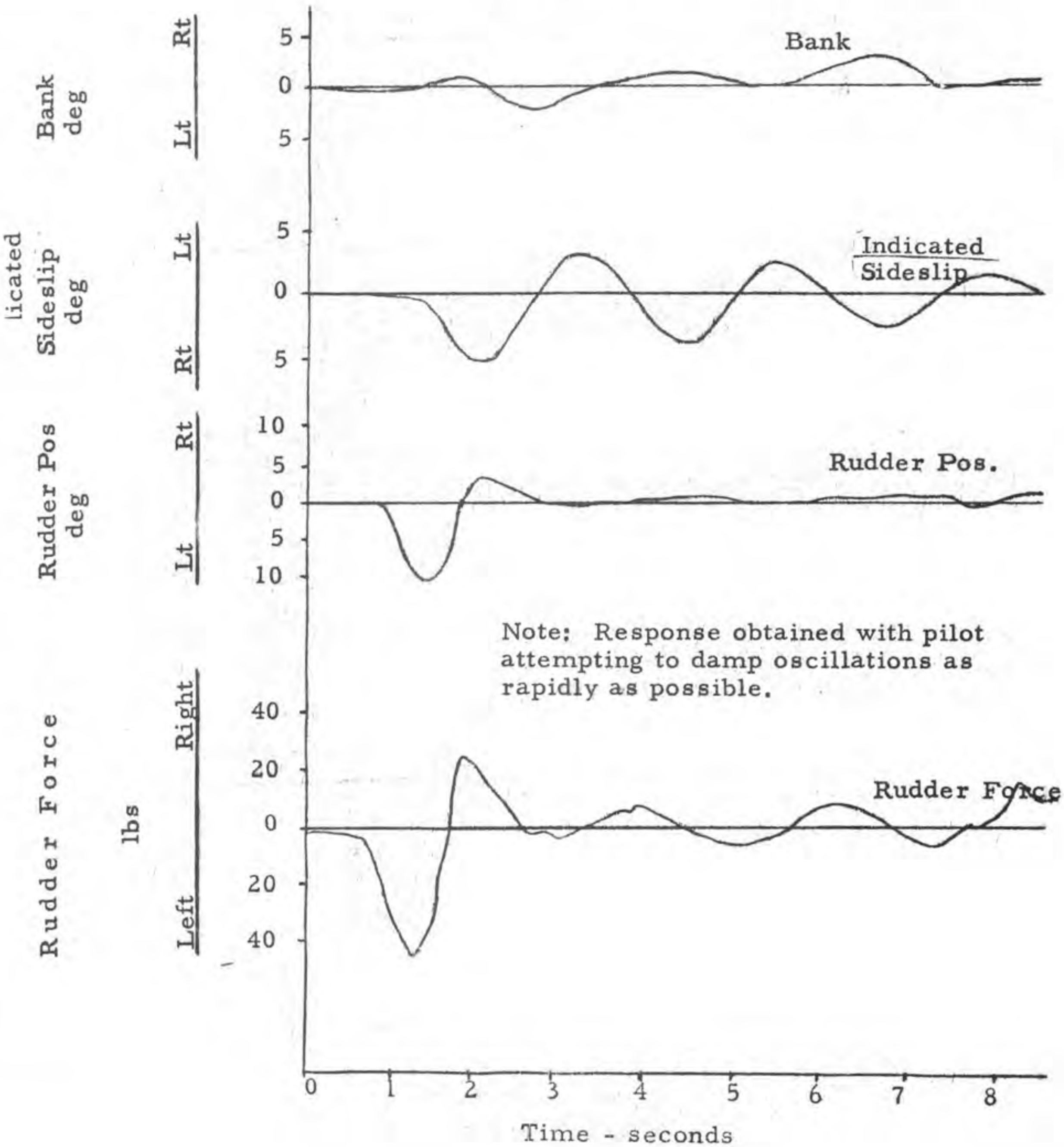


Figure No. 68

DYNAMIC DIRECTIONAL STABILITY
YF-100A, USAF No. 52-5754
Power Configuration (A/B Off) Controls Free

TRIM CONDITIONS

CAS 475 knots — Altitude 10,500 feet
CG 31.1 o/o MAC — Weight 22,900 lbs
Ave N_2 9550 RPM — Rudder Pos. 0.5 degrees Rt
L. Aileron Position 0 degrees
No external stores attached

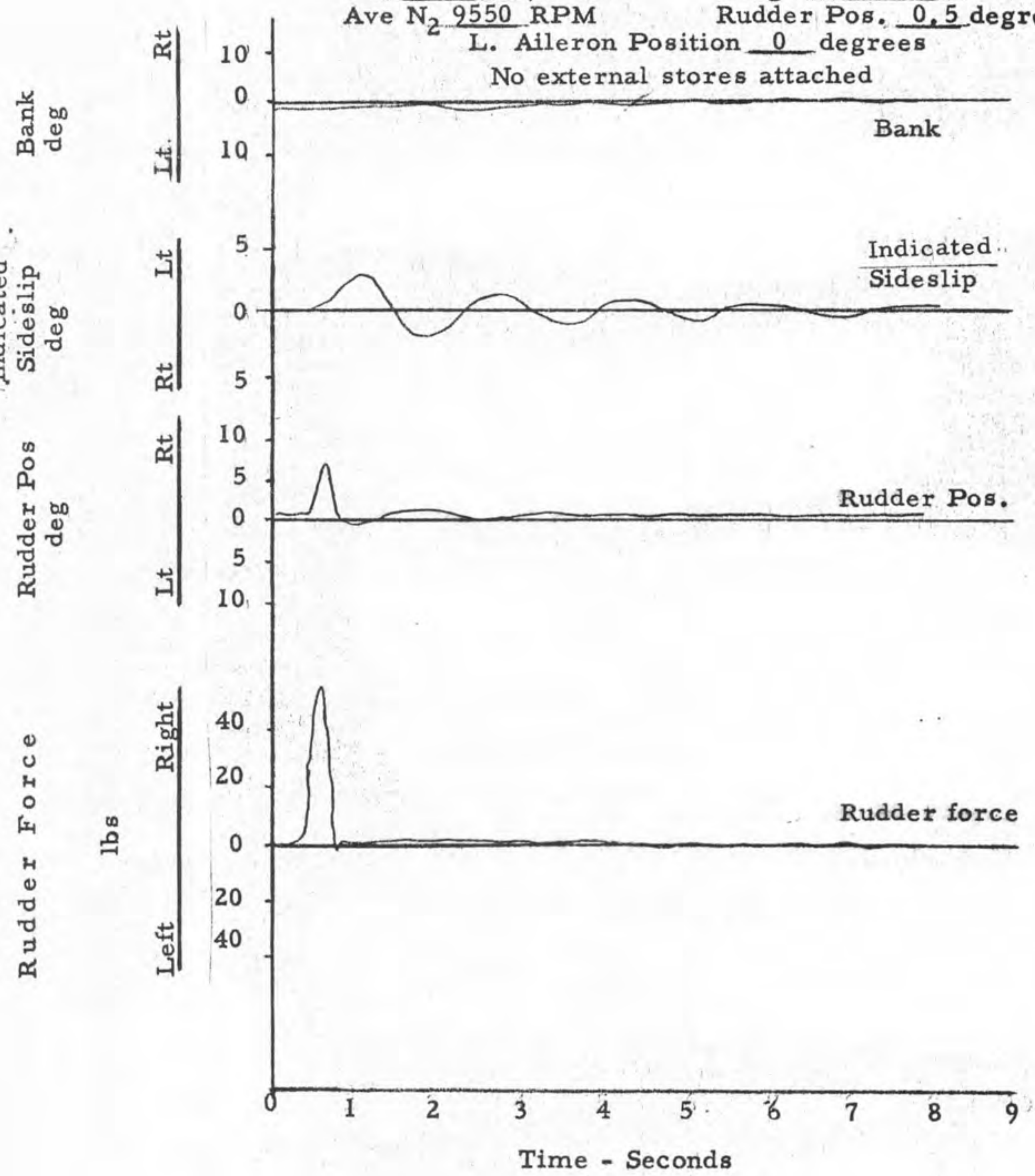


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Figure No. 69

DYNAMIC DIRECTIONAL STABILITY
YF-100A, USAF No. 52-5754
Power Configuration (A/B Off) Controls Free

TRIM CONDITIONS

CAS 475 knots — Altitude 10,000 feet
CG 31.1 °/o MAC — Weight 22,900 lbs
Ave N_2 9550 RPM — Rudder Pos. 0.5 degrees Rt
L. Aileron Pos. 0 degrees

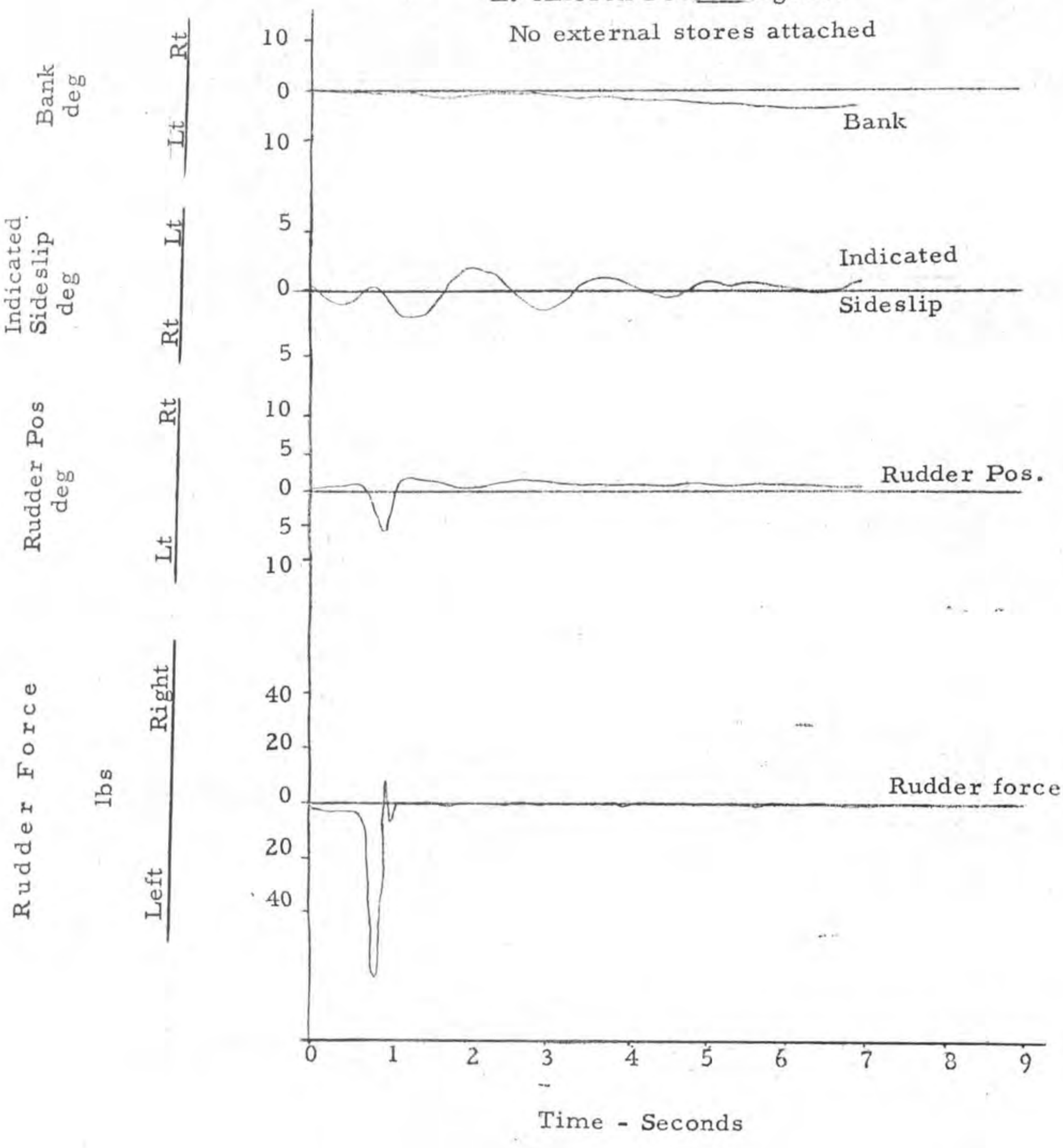

Figure No. 70

DYNAMIC DIRECTIONAL STABILITY
YF-100 A, USAF No. 52-5754
Power Configuration (A/B off)

TRIM CONDITIONS

CAS 475 knots Altitude 10,500 feet
CG 31.1 o/o MAC Weight 22,900 lbs
Ave N_2 9550 RPM Rudder Pos. 0.5 degrees Rt
L. Aileron Position 0 degrees

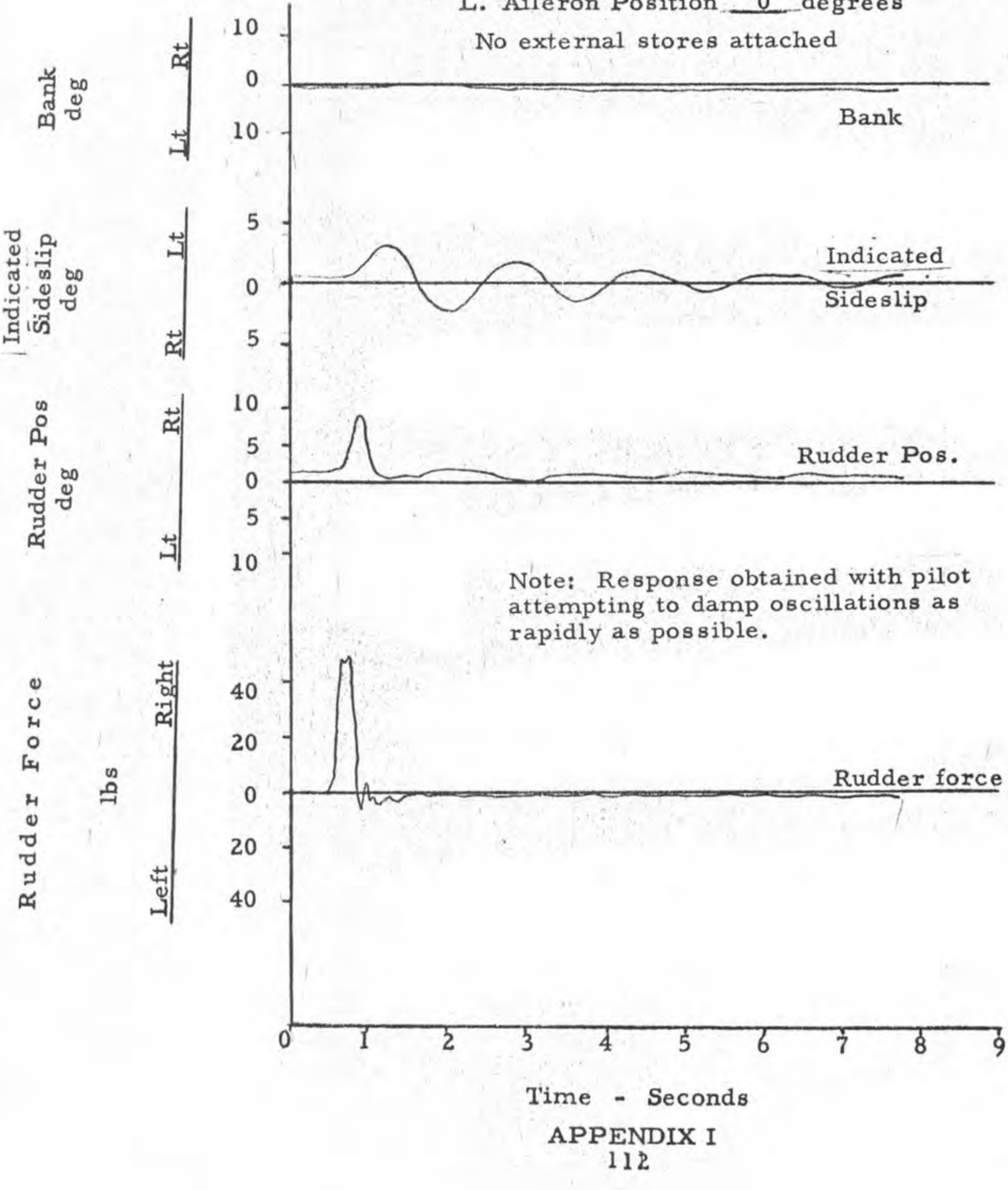

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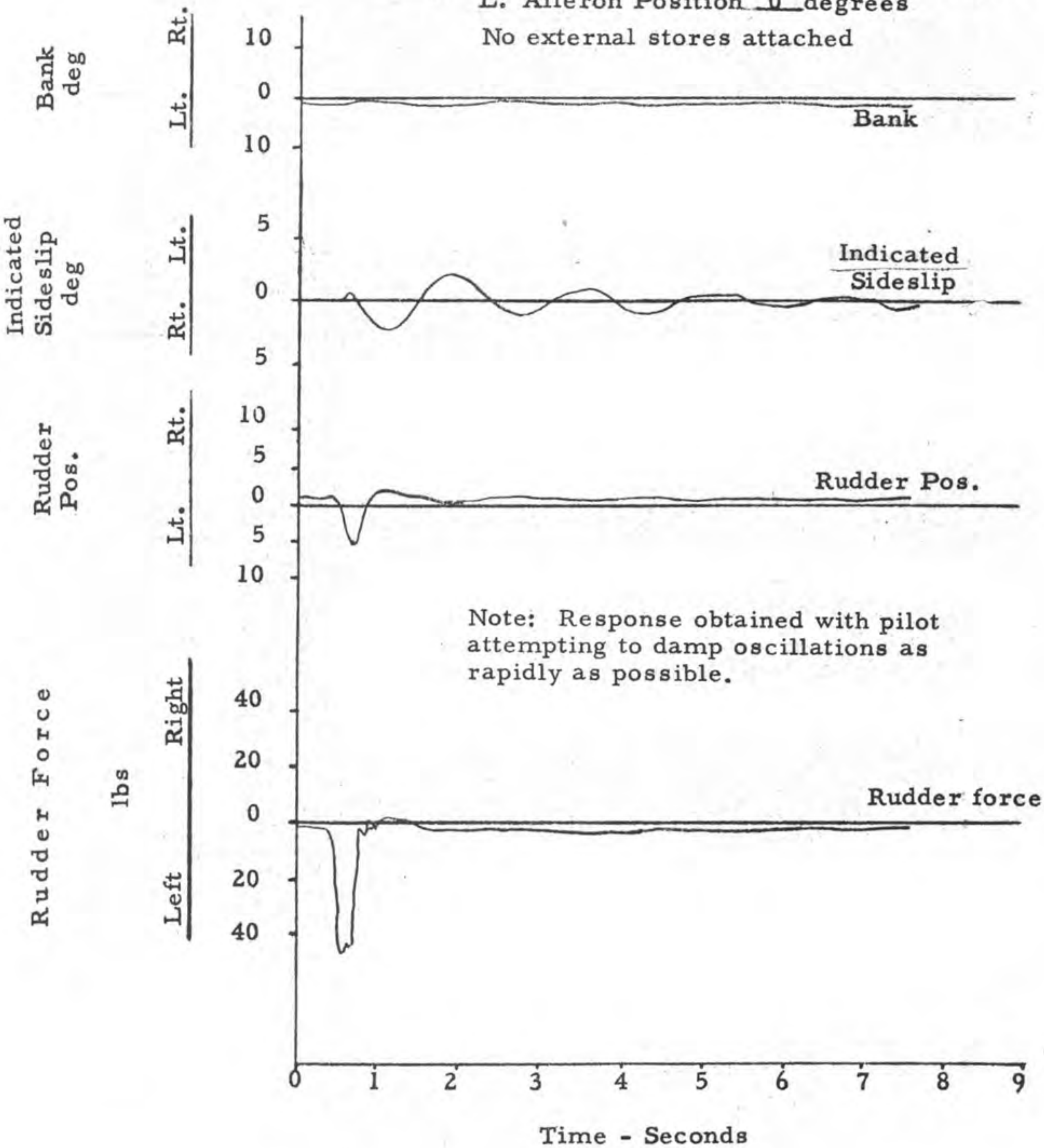
Figure No. 71

DYNAMIC DIRECTIONAL STABILITY
YF-100 A, USAF No. 52-5754
Power Configuration (A/B off)

TRIM CONDITIONS

CAS 475 knots — Altitude 10,500 feet
CG 31.1 % MAC — Weight 22,900 lbs
Ave N_2 9550 RPM — Rudder Pos. 0.5 degrees Rt
L. Aileron Position 0 degrees
No external stores attached



APPENDIX I

Figure No. 72

DYNAMIC DIRECTIONAL STABILITY
YF-100A. USAF No. 52-5754
Power Approach Configuration Controls Free

TRIM CONDITIONS

IAS 164 knots Altitude 9,900 feet
CG 30.4 o/o MAC Weight 21,300 lbs
Ave N_2 9240 RPM Rudder Pos. 2.3 degrees Rt.
L. Aileron Position 0.5 degrees Up

No external Stores attached

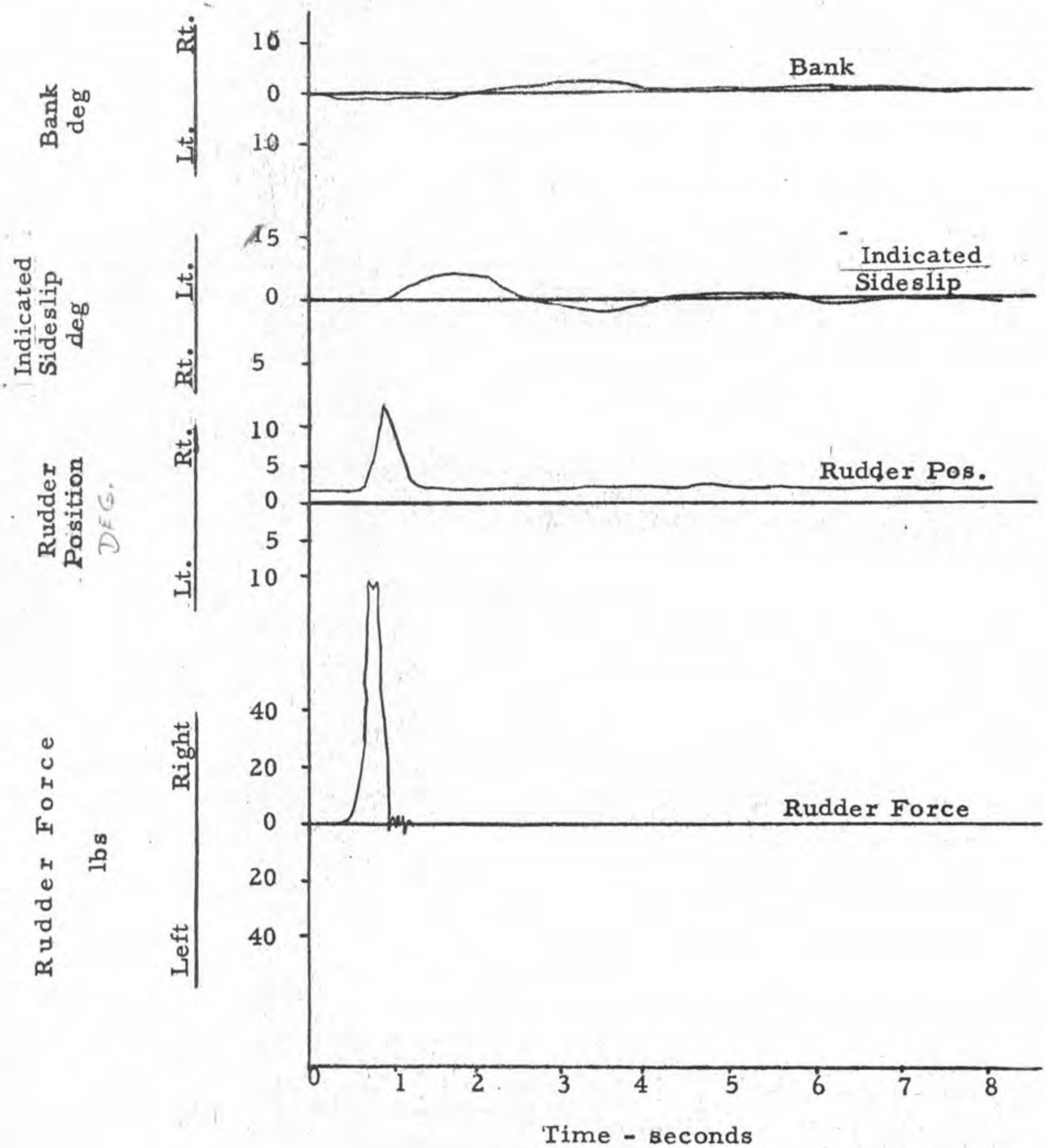


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Figure No.73

DYNAMIC DIRECTIONAL STABILITY
YF-100 A, USAF No. 52-5754
Power Approach Configuration Controls Free

TRIM CONDITIONS

IAS 164 knots — Altitude 9,900 feet
CG 30.4 o/o MAC — Weight 21,300 lbs
Ave N_2 9240 RPM — Rudder Pos. 2.3 degrees Rt
L. Aileron Position 0.5 degrees Up

No external stores attached

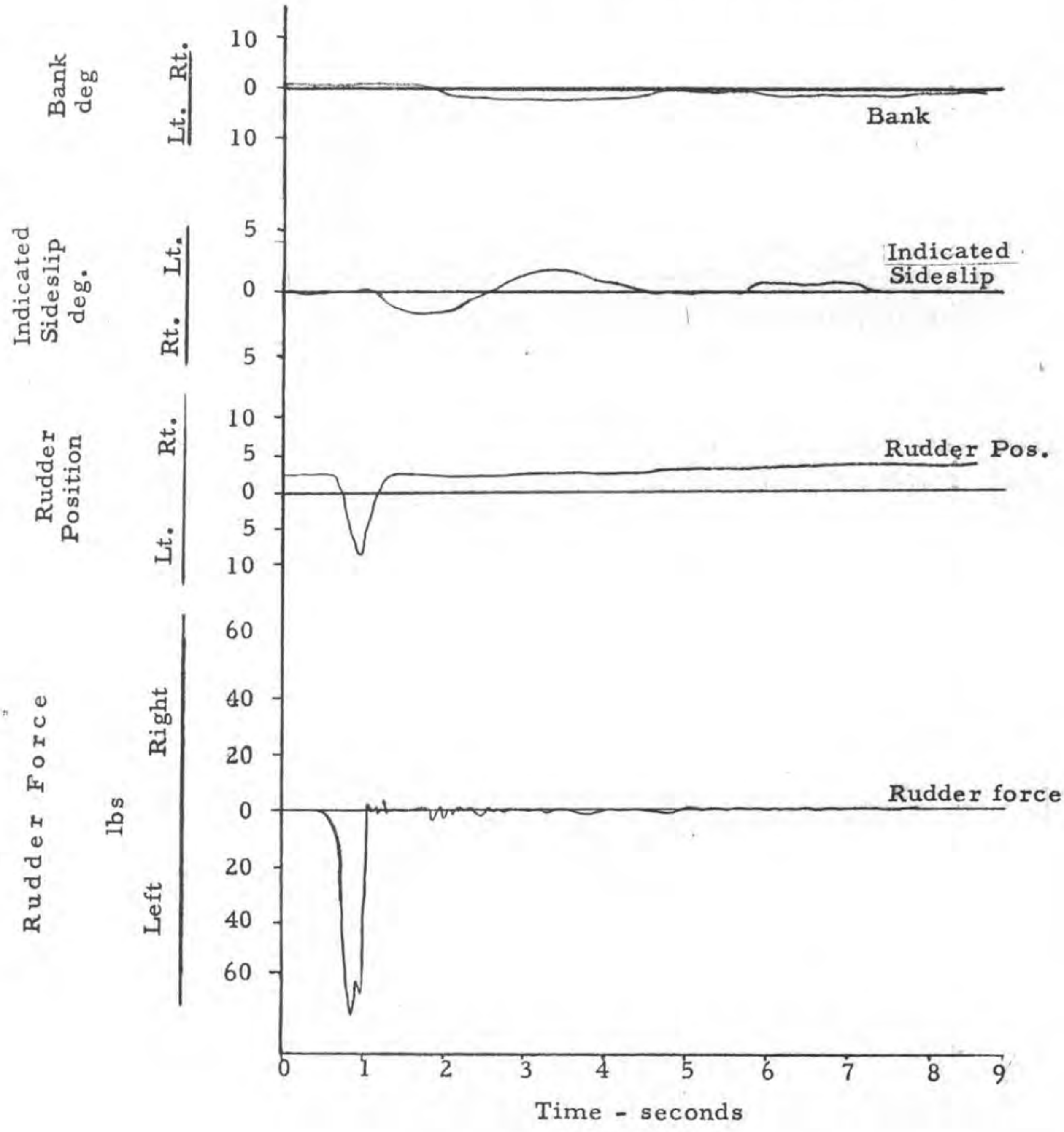

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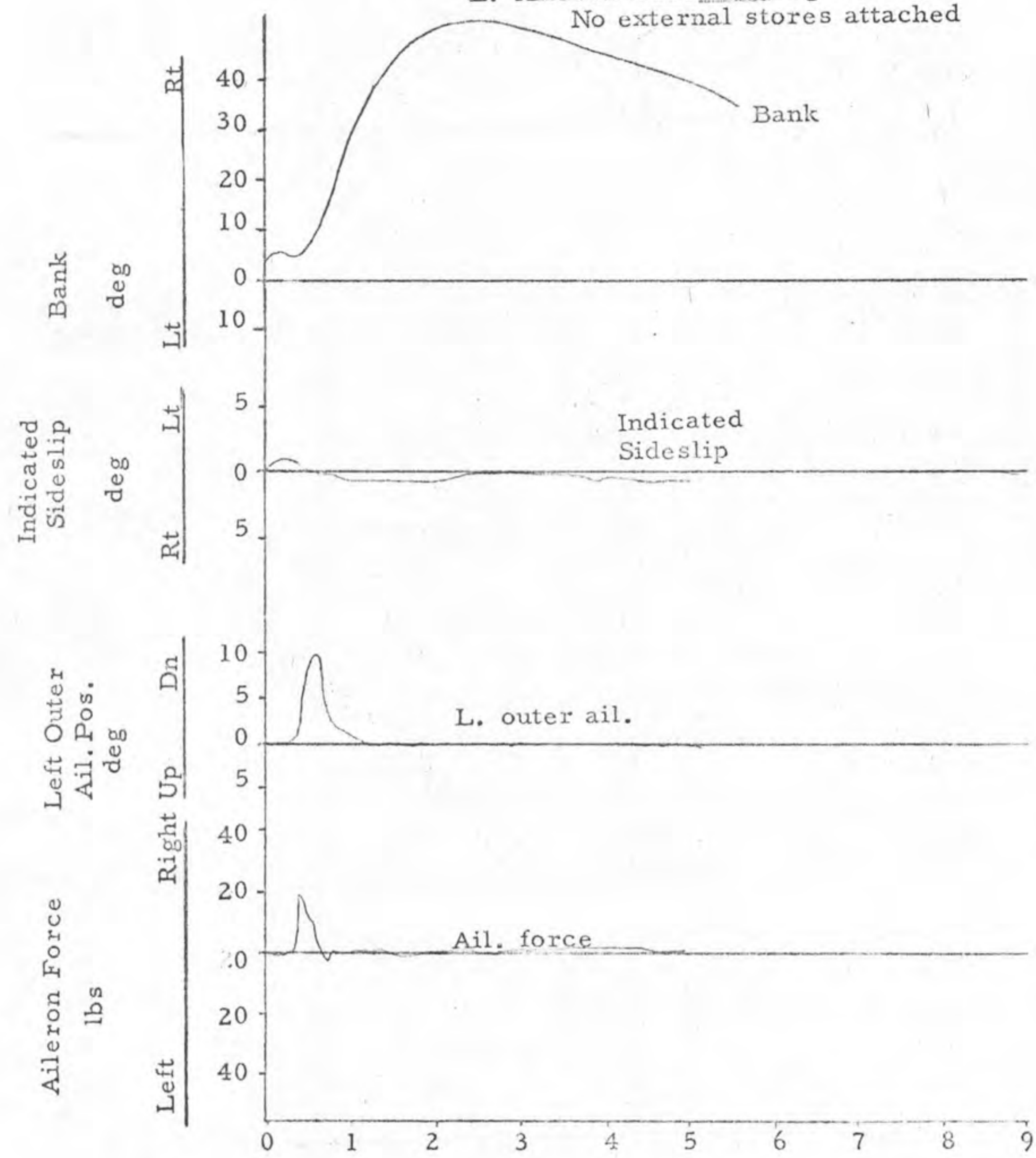
Figure No. 74

DYNAMIC LATERAL STABILITY
YF-100 A, USAF No. 52-5754

Power Configuration (A/B On) Controls Free

TRIM CONDITIONS

CAS 274 knots — Altitude 45,300 feet
CG 29.5 °/oMAC — Weight 21,800 lbs
Ave N_2 9070 RPM — Rudder Pos. 0 deg.
L. Aileron Pos. 0.4 Degrees Up
No external stores attached



APPENDIX I

Figure No. 75

DYNAMIC LATERAL STABILITY
YF-100A, USAF No. 52-5754
Power Configuration (A/B On) Controls Free

TRIM CONDITIONS

CAS 274	knots	Altitude 45,300 feet	
CG 29.5	o/o MAC	Weight 21,800 lbs	
Ave N_2 9070	RPM	Rudder Pos. 0 deg	

L. Aileron Position 0.4 degrees Up

No external stores attached

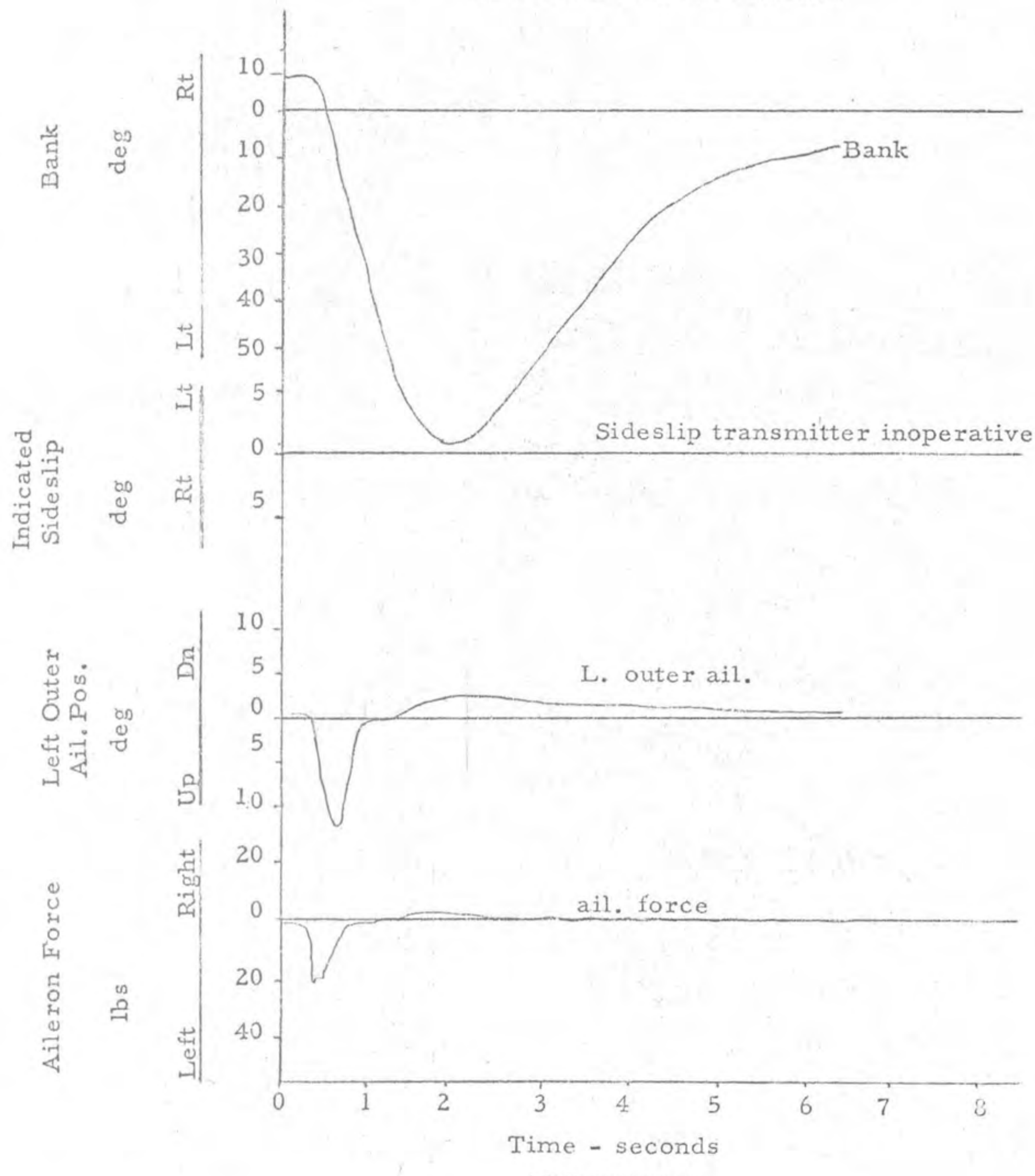


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Figure No. 76

DYNAMIC LATERAL STABILITY
YF-100A, USAF No. 52-5754
Power Configuration (A/B on)

TRIM CONDITIONS

CAS 274 knots
CG 29.5 o/o MAC
Ave N_2 9070 RPM

Altitude 45,300 feet
Weight 21,800 lbs
Rudder Pos. 0 deg

L Aileron Position 0.4 degrees Up
No external stores attached

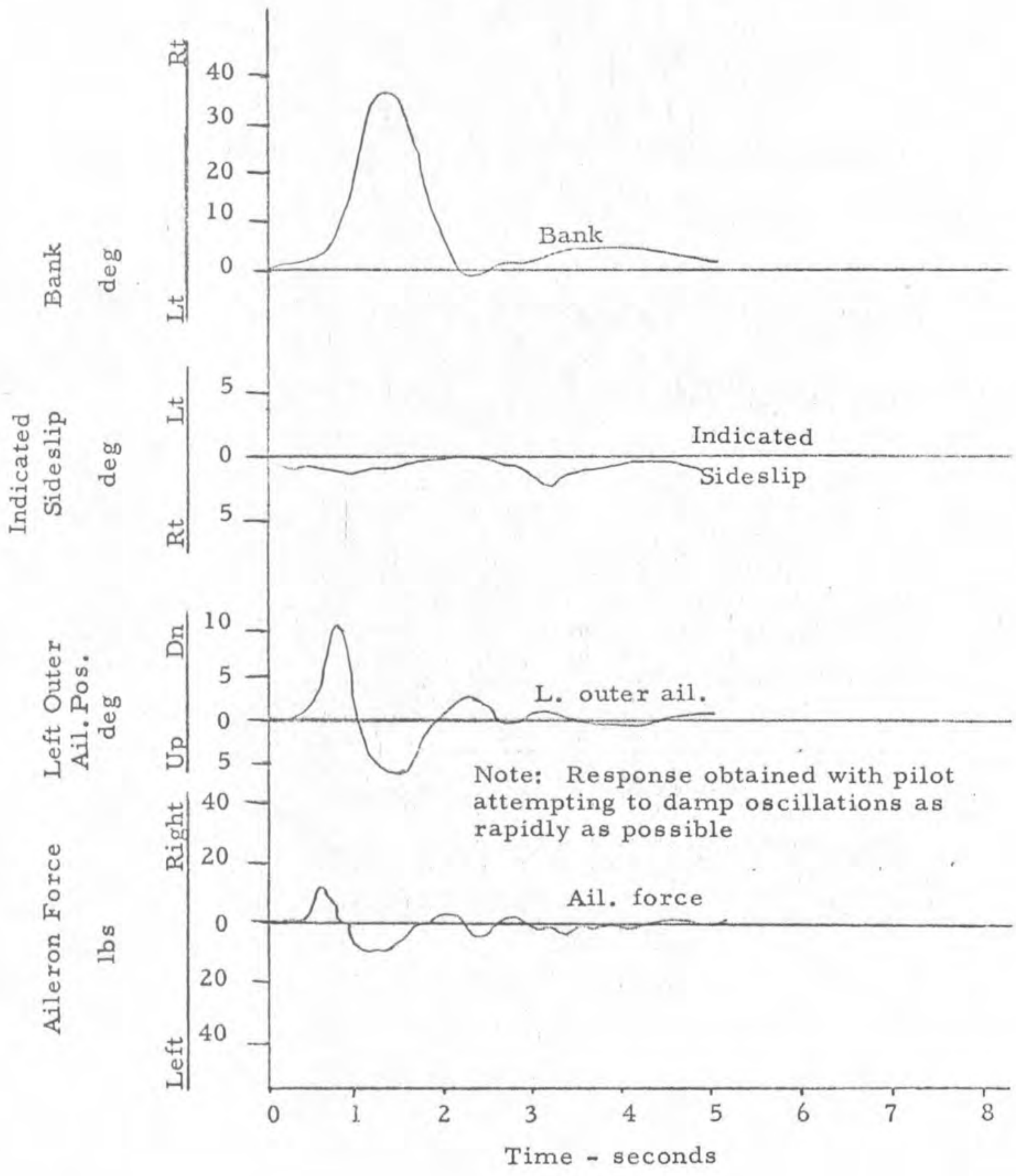

Figure No. 77

DYNAMIC LATERAL STABILITY
YF-100A, USAF No. 52-5754
Power Configuration (A/B on)

TRIM CONDITIONS

CAS 274 knots
CG 29.5 o/o MAC
Ave N_2 9070 RPM
Altitude 45,300 feet
Weight 21,800 lbs
Rudder Pos. 0 deg
L. Aileron Position 0.4 degrees Up

No external stores attached

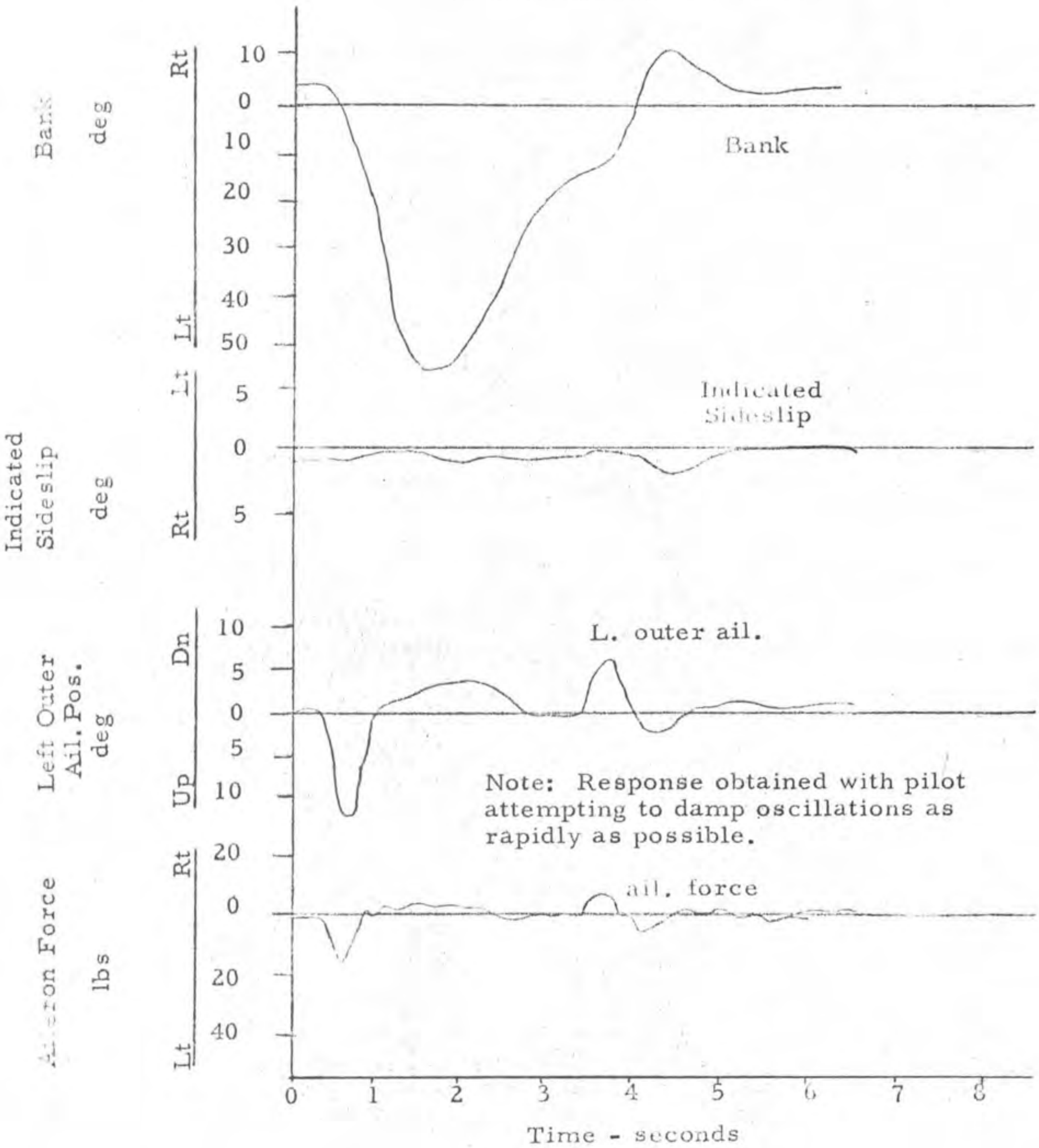

Figure No. 78

DYNAMIC LATERAL STABILITY
YF-100A, USAF No. 52-5754
Power Configuration (A/B Off) Controls Free

TRIM CONDITIONS

CAS 475 knots | Altitude 10,500 feet
CG 30.5 % MAC | Weight 22,500 lbs
Ave N_2 9550 RPM | Rudder Pos. 0.5 degrees Rt

L. Aileron Pos. 0 degrees
No external stores attached

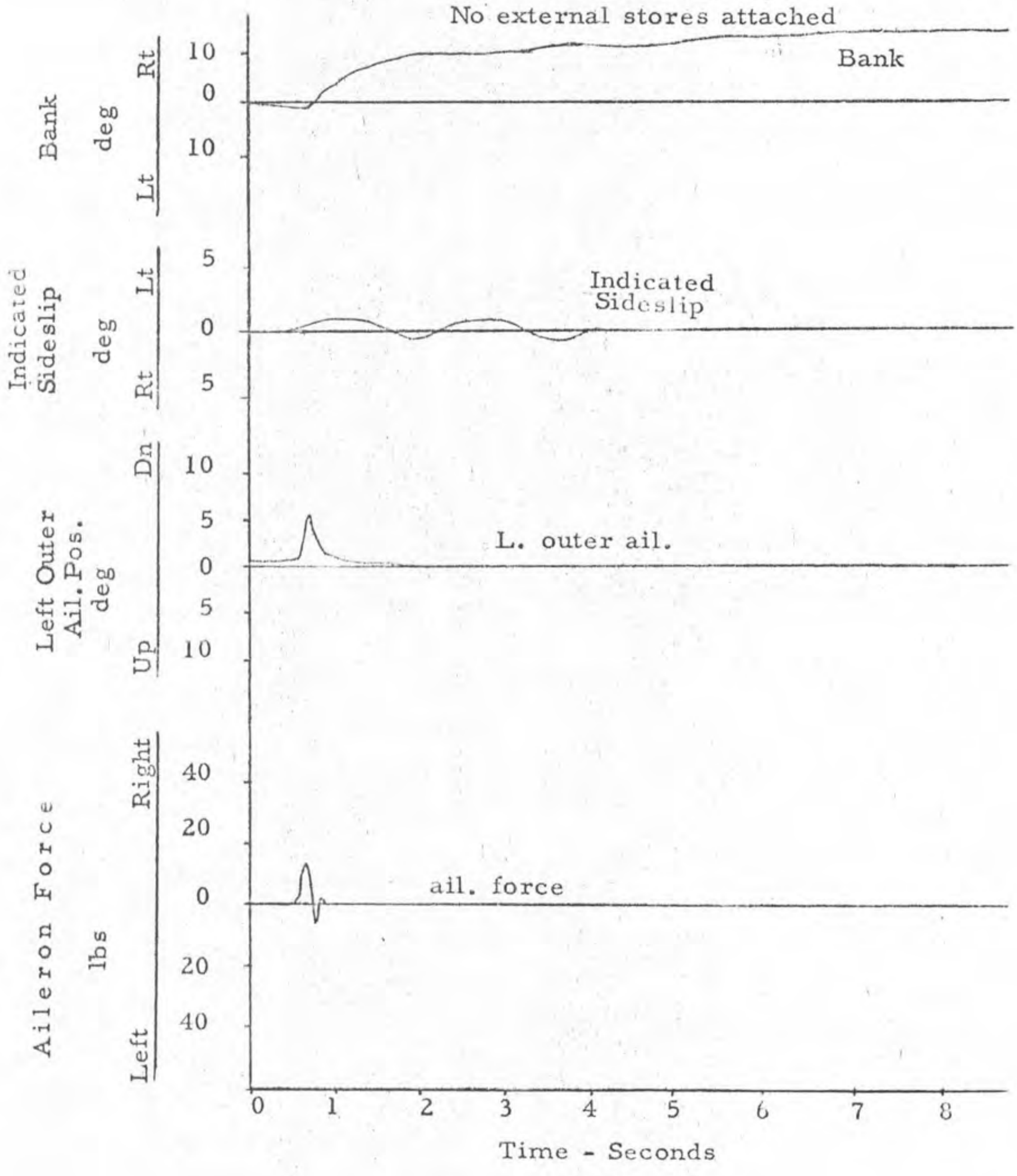

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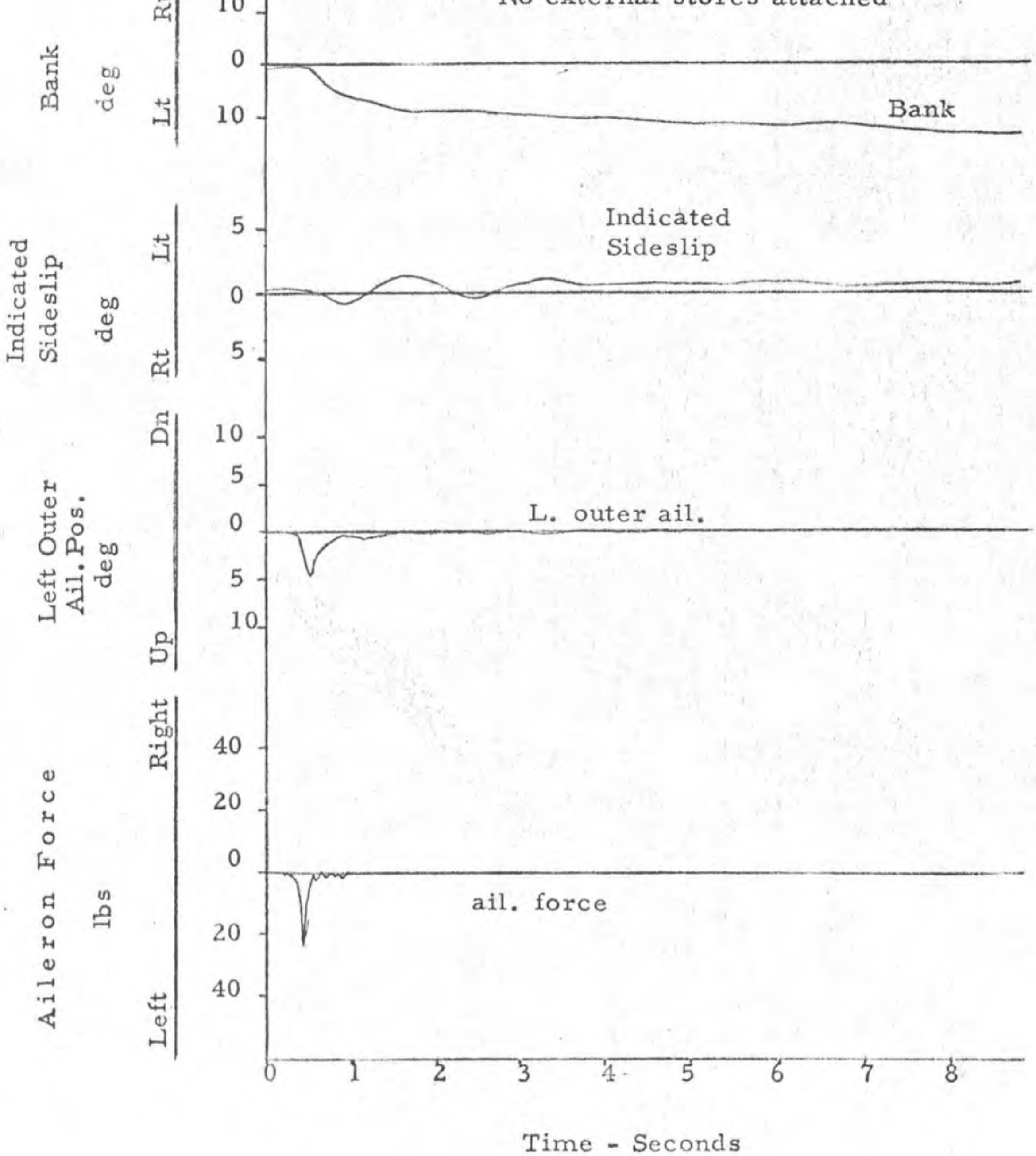
Figure No. 79

DYNAMIC LATERAL STABILITY
YF-100A, USAF No. 52-5754
Power Configuration (A/B Off) Controls Free

TRIM CONDITIONS

CAS 475 knots
CG 30.5 °/o MAC
Ave N_2 9550 RPM
Altitude 10,500 feet
Weight 22,500 lbs
Rudder Pos. 0.5 degrees Rt
L. Aileron Pos. 0 degrees
No external stores attached



Time - Seconds

Figure No. 80

DYNAMIC LATERAL STABILITY
YF-100A, USAF No. 52-5754
Power Configuration (A/B off)

TRIM CONDITIONS

CAS 475 knots Altitude 10,500 feet
CG 30.5 ^{o}/o MAC Weight 22,500 lbs
Ave N_2 9550 RPM Rudder Pos. 0.5 Degrees Rt
L. Aileron Position 0 degrees
No external stores attached

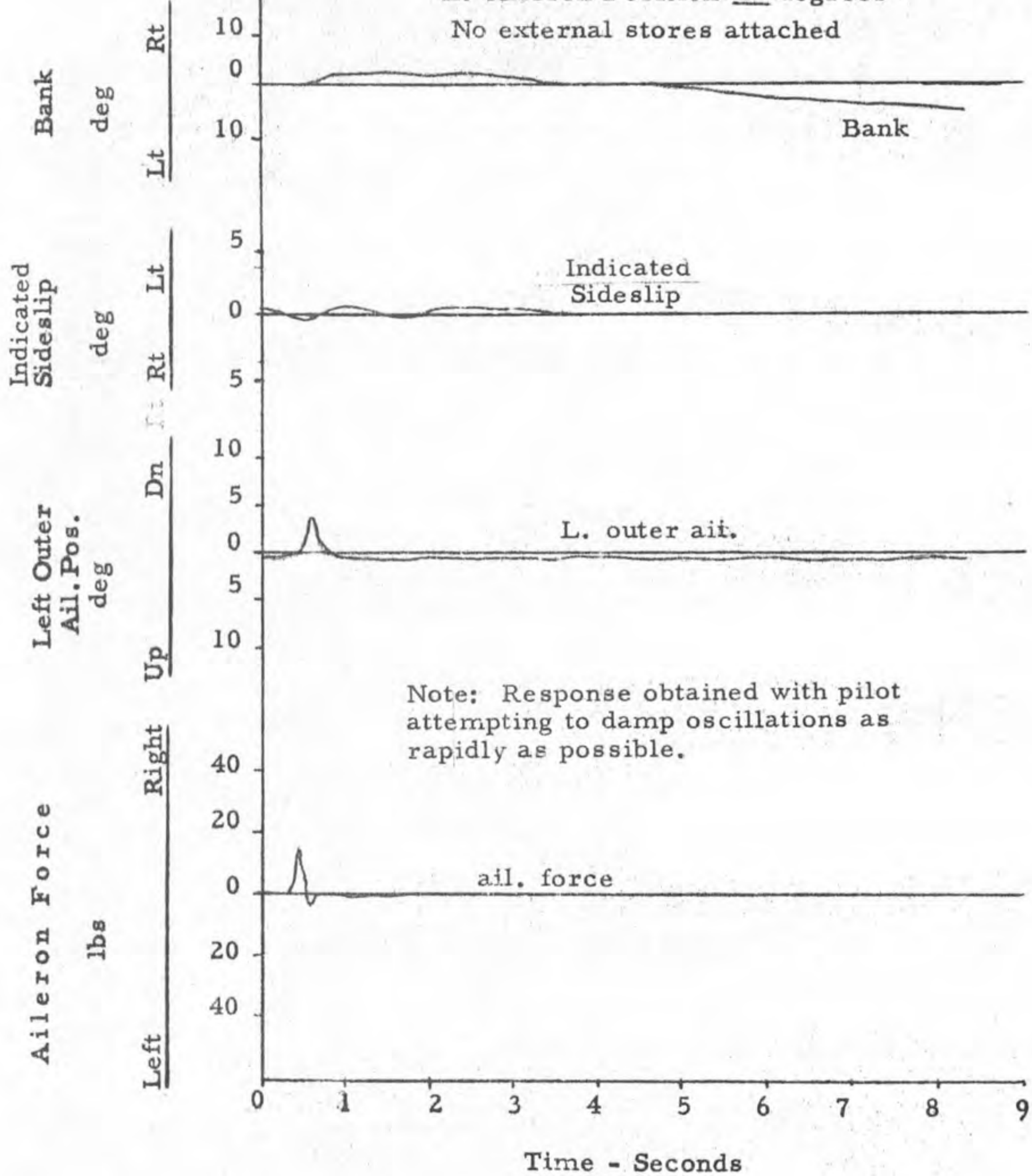


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Figure No. 81

DYNAMIC LATERAL STABILITY
YF-100 A, USAF No. 52-5754
Power Configuration (A/B off)

TRIM CONDITIONS

CAS 475 knots Altitude 10,500 feet
CG 30.5 % MAC Weight 22,500 lbs
Ave N_2 9550 RPM Rudder Pos. 0.5 degrees Rt
L. Aileron Pos. 0 degrees

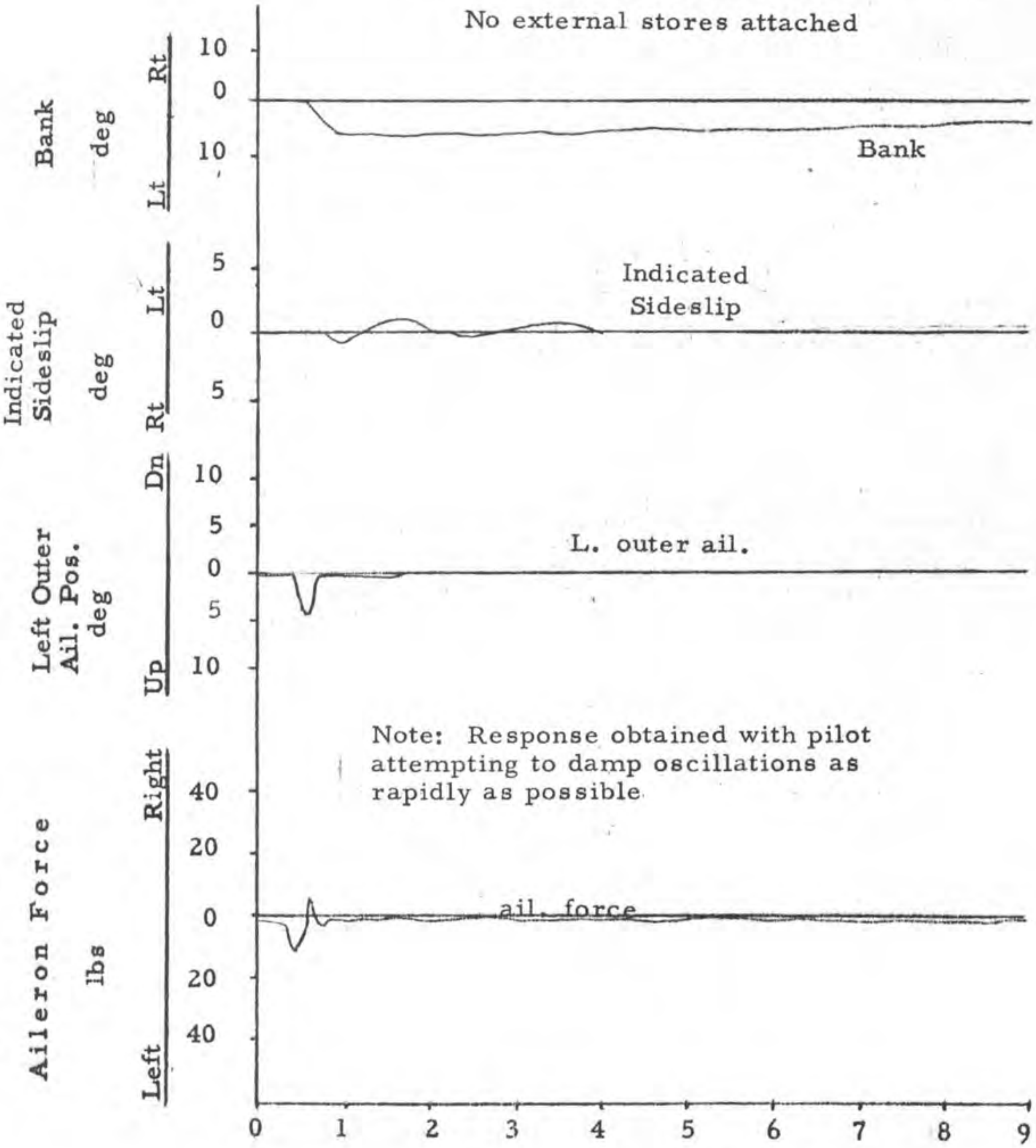

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Figure No. 82

DYNAMIC LATERAL STABILITY
YF-100A, USAF No. 52-5754
Power Approach Configuration Controls Free

TRIM CONDITIONS

IAS 164 knots Altitude 9,900 feet
CG 30.5 o/o MAC Weight 21,200 lbs
Ave N_2 9240 RPM Rudder Pos. 2.3 degrees Rt
L. Aileron Position 0.5 degrees Up

No external stores attached

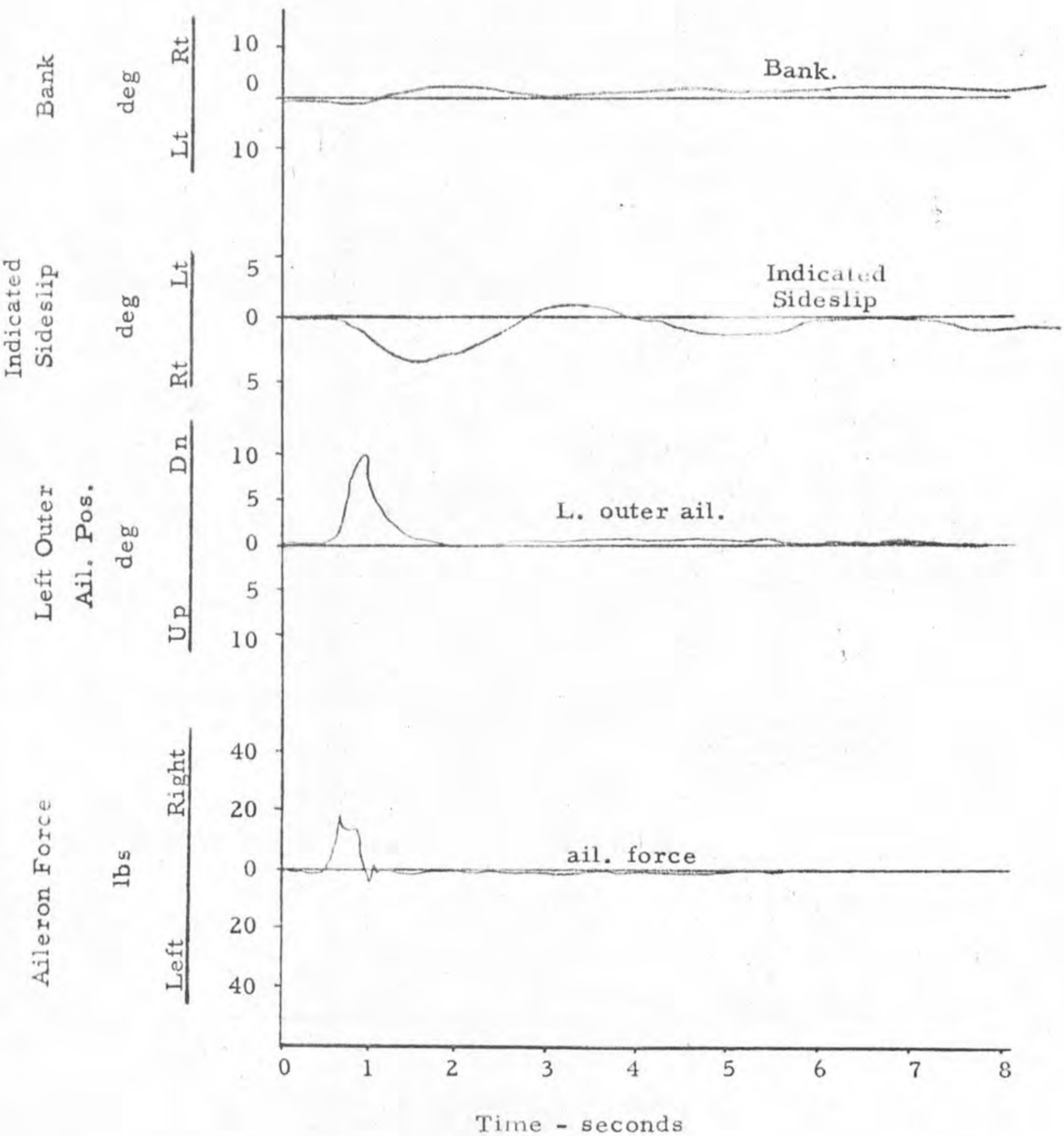

Figure No. 83

DYNAMIC LATERAL STABILITY
YF-100A, USAF No. 52-5754
Power Approach Configuration Controls Free

TRIM CONDITIONS

IAS 164 knots Altitude 9,900 feet
CG 30.5 °/o MAC Weight 21,200 lbs
Ave N_2 9240 RPM Rudder Pos. 2.3 degrees Rt
L. Aileron Position 0.5 degrees Up

No external stores attached

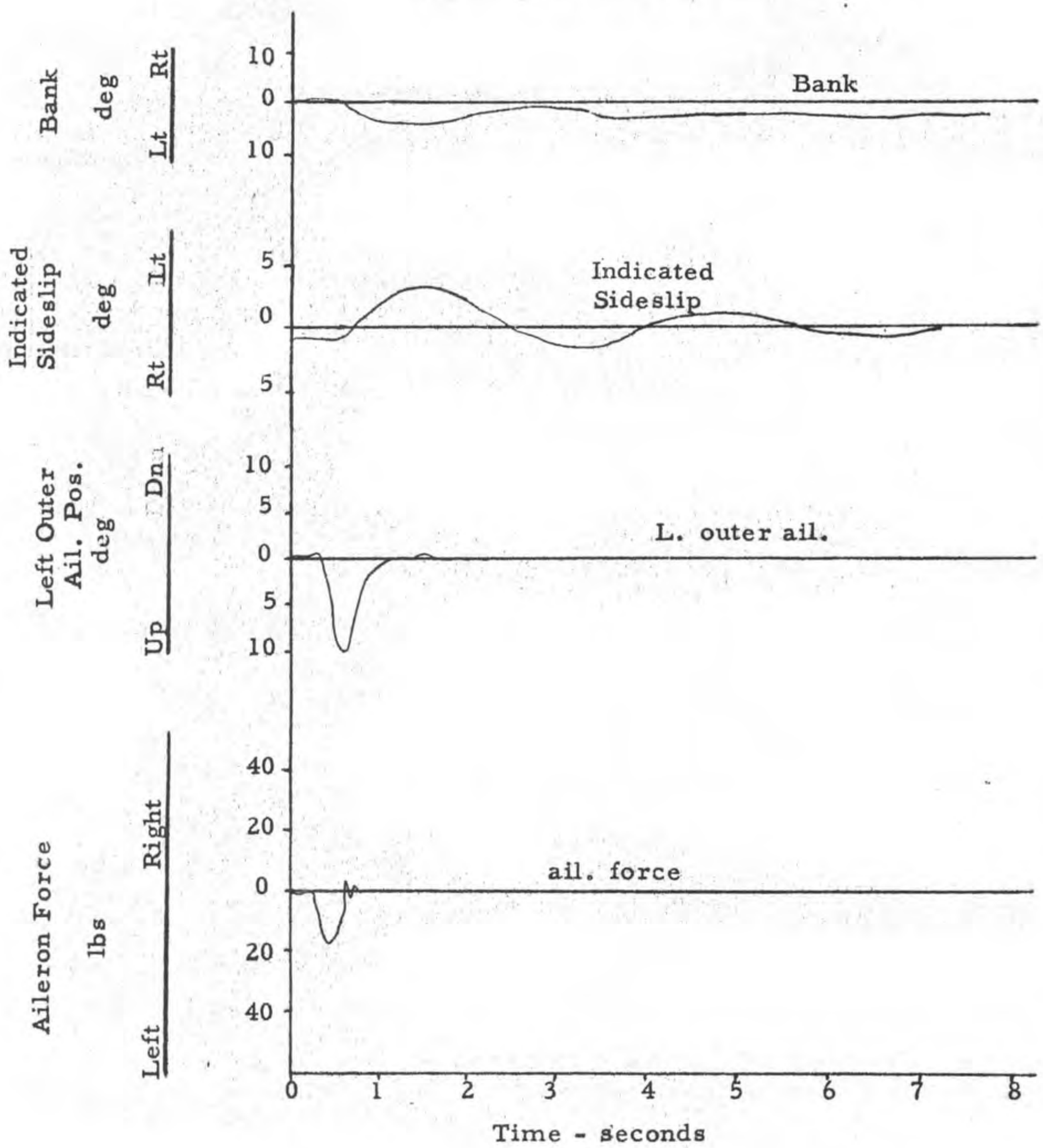

Figure No. 84

ADVERSE YAW
YF-100A, USAF No. 52-5754
Power Approach Config.

TRIM CONDITIONS

IAS 167 knots — Altitude 10,400 feet
CG 31.4 %MAC — Weight 23,000 lbs
Rudder Pos. 2° left — L. Outer Ail. Pos. 0

No external stores attached

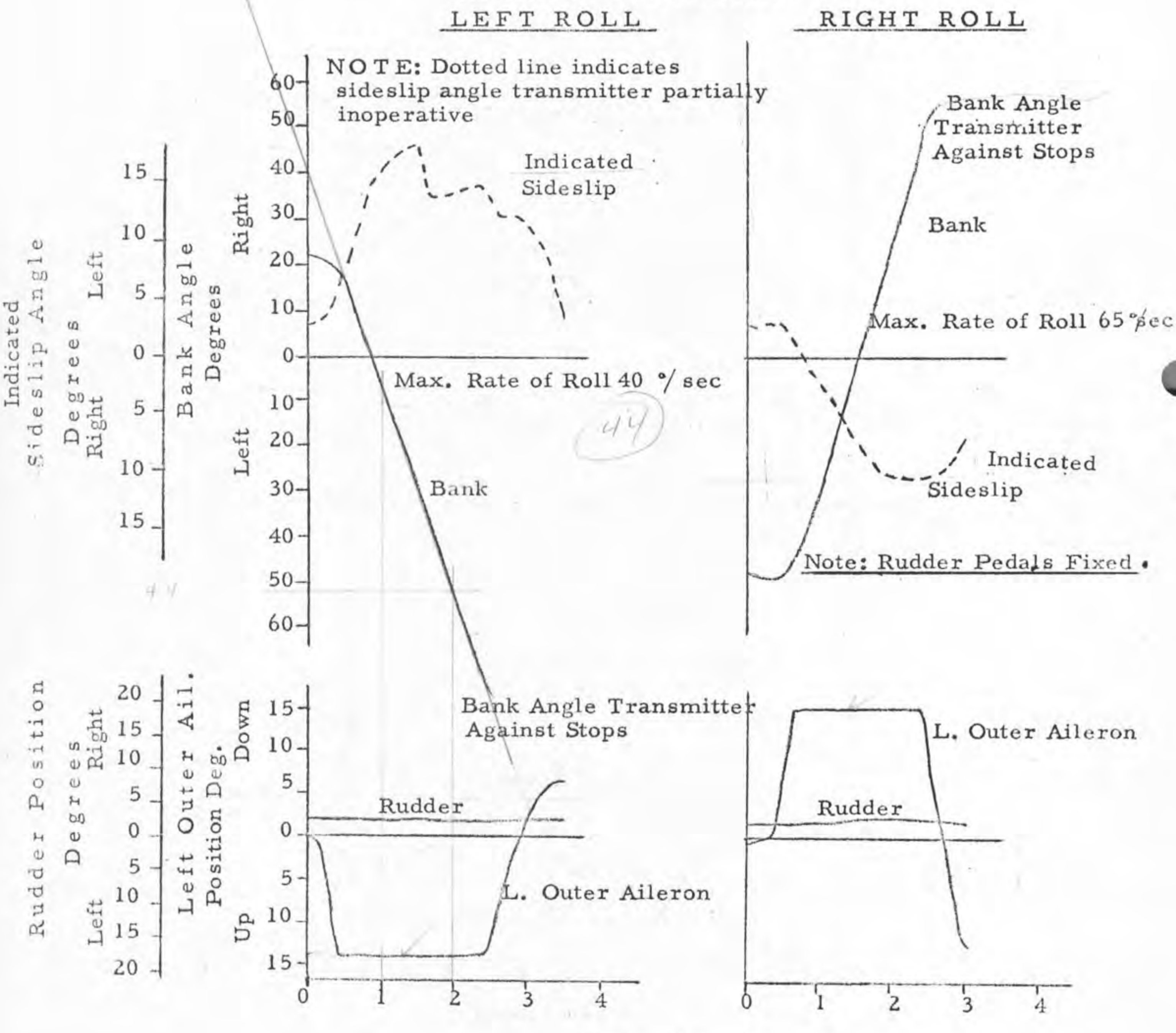

Figure No. 85

ADVERSE YAW
YF-100A, USAF No. 52-5754
Power Approach Configuration

TRIM CONDITIONS

IAS 167 knots — Altitude 10,400 feet
CG 31.4 %MAC — Weight 23,000 lbs
Rudder Pos. 2° left — L. Outer Ail. Pos. 0
No external stores attached

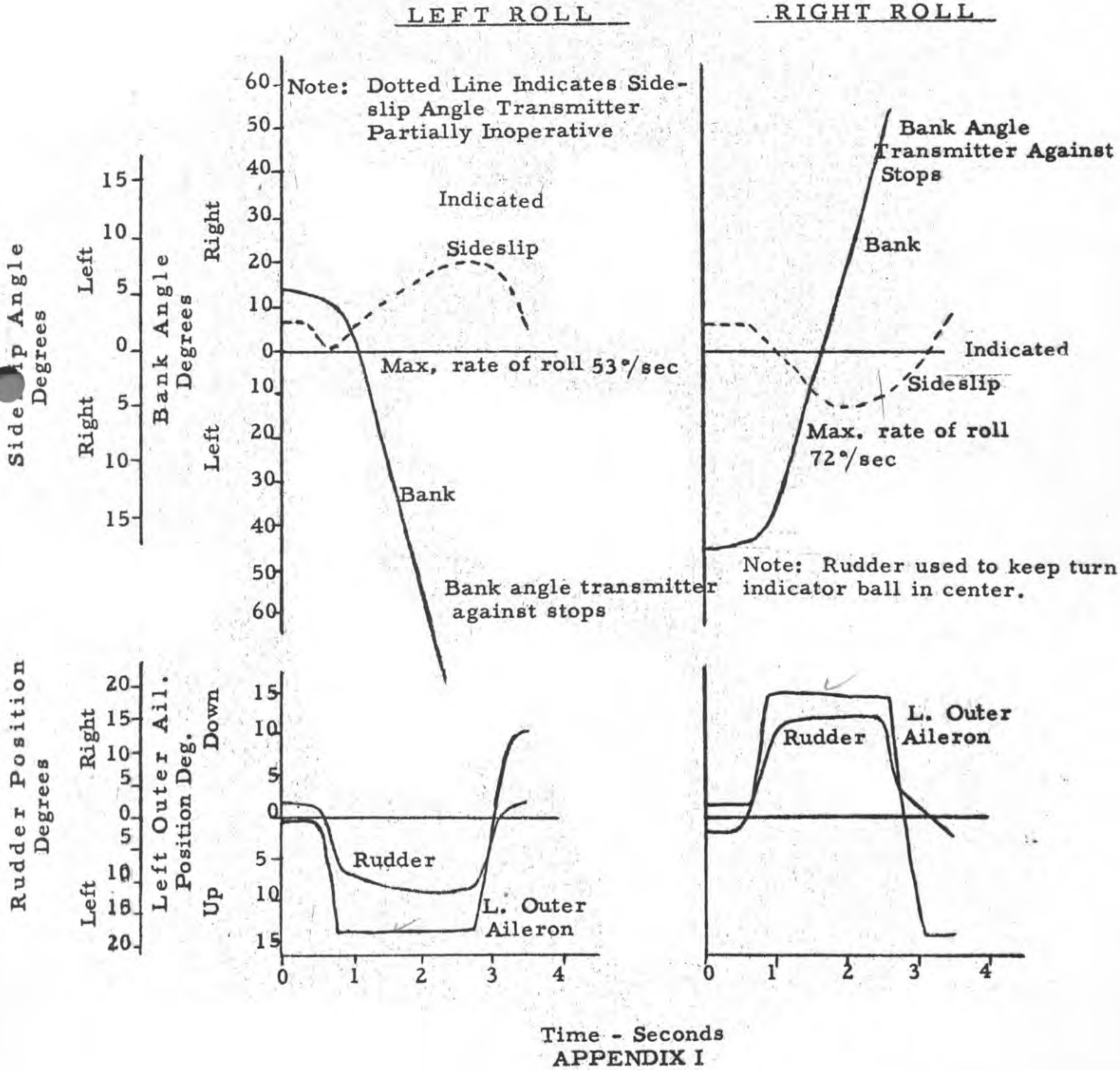

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Figure No. 86

ADVERSE YAW

YF-100A, USAF No. 52-5754

Power Configuration(A/B Off)

TRIM CONDITIONS

CAS 245 knots | Altitude 45000 feet

CG 30.6 ^{o}/o MAC | Weight 22,500 lbs

Rudder Pos. 2° left | L. Outer Ail. Pos. .5° Dn

No external stores attached

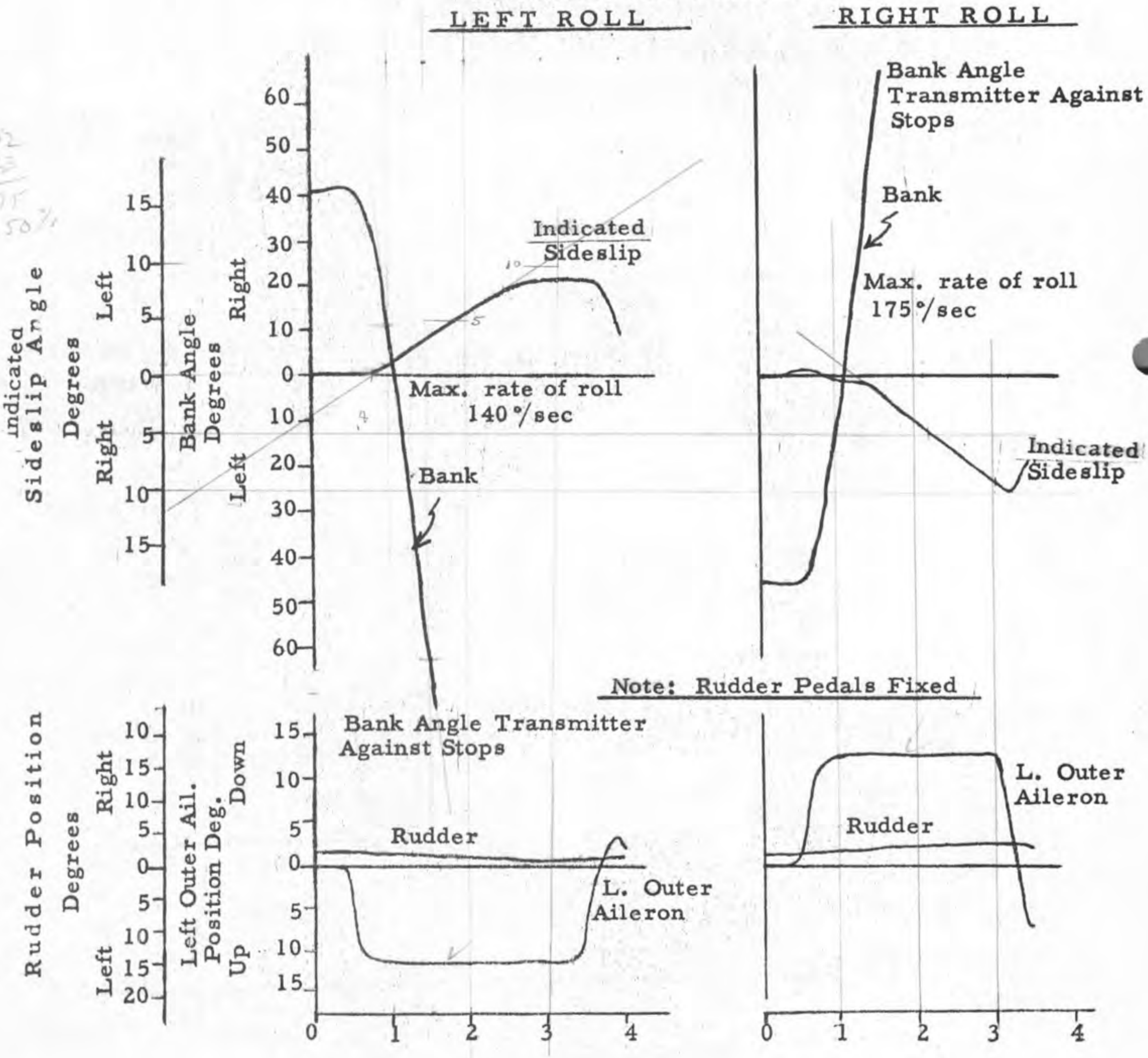

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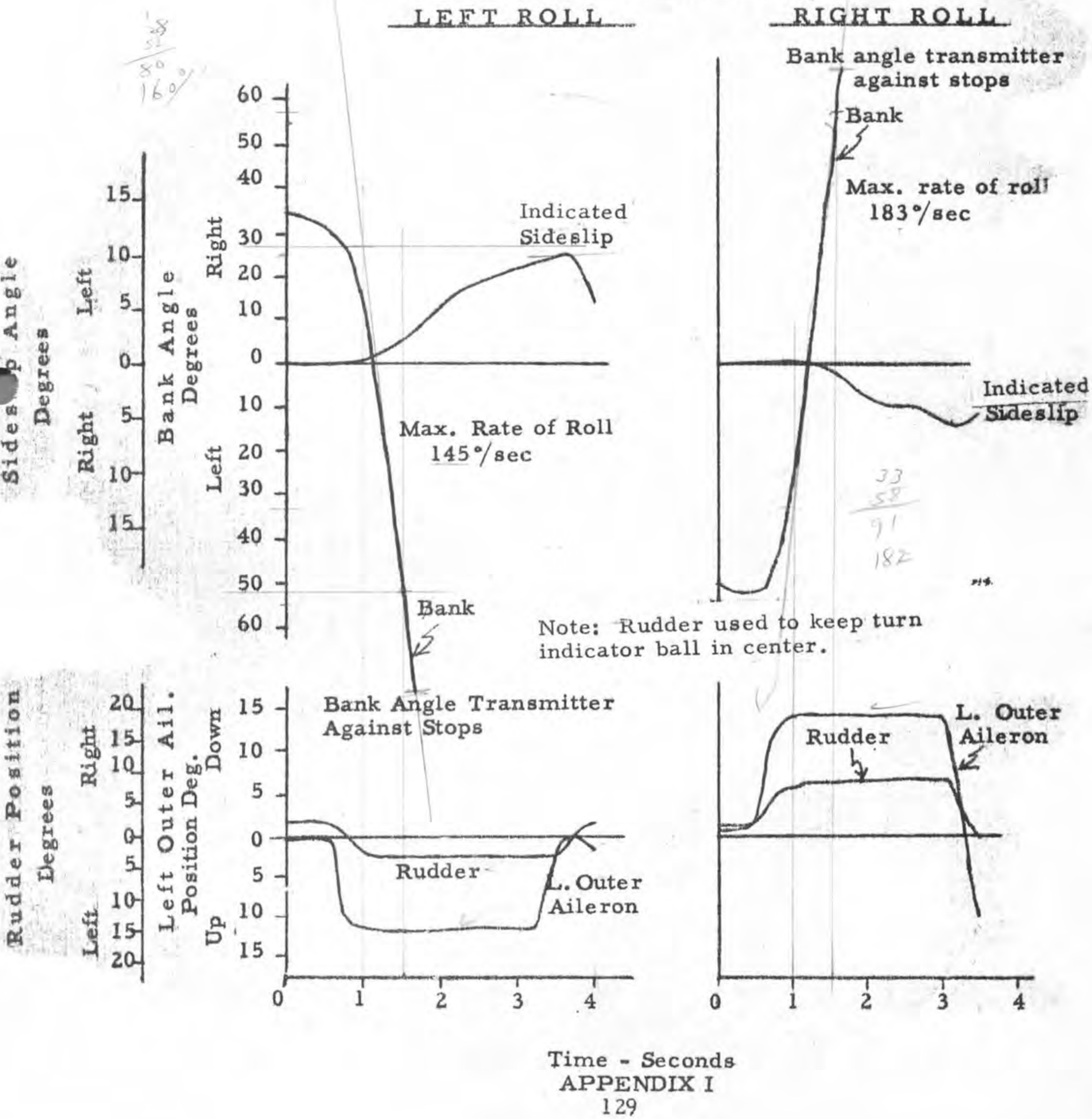
Figure No. 87

ADVERSE YAW
YF-100A, USAF No. 52-5754
Power Configuration (A/B Off)

TRIM CONDITIONS

CAS 244 knots — Altitude 45000 feet
CG 30.6 %MAC — Weight 22500 lbs
Rudder Pos. 2° left — L. Outer Ail. Pos. .5° Dn
No external stores attached



Time - Seconds

FIG NO 88
SUMMARY OF ROLLING CHARACTERISTICS
YF-100A USAF No 52-5754
POWER CONFIGURATION
POWER FOR LEVEL FLIGHT
RUDDER PEDALS FIXED
ALTITUDE ~ 45000 FEET
NO EXTERNAL STORES ATTACHED

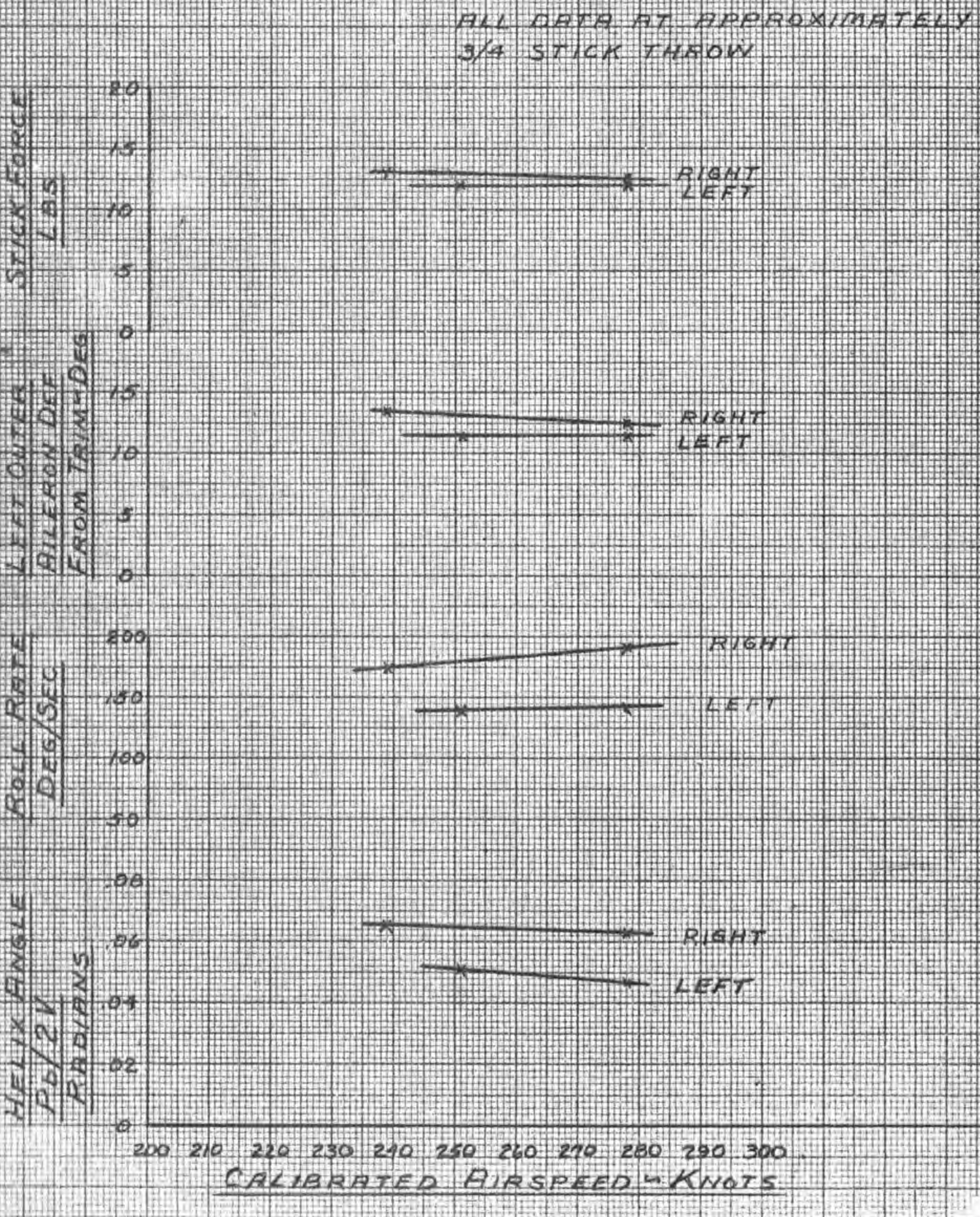
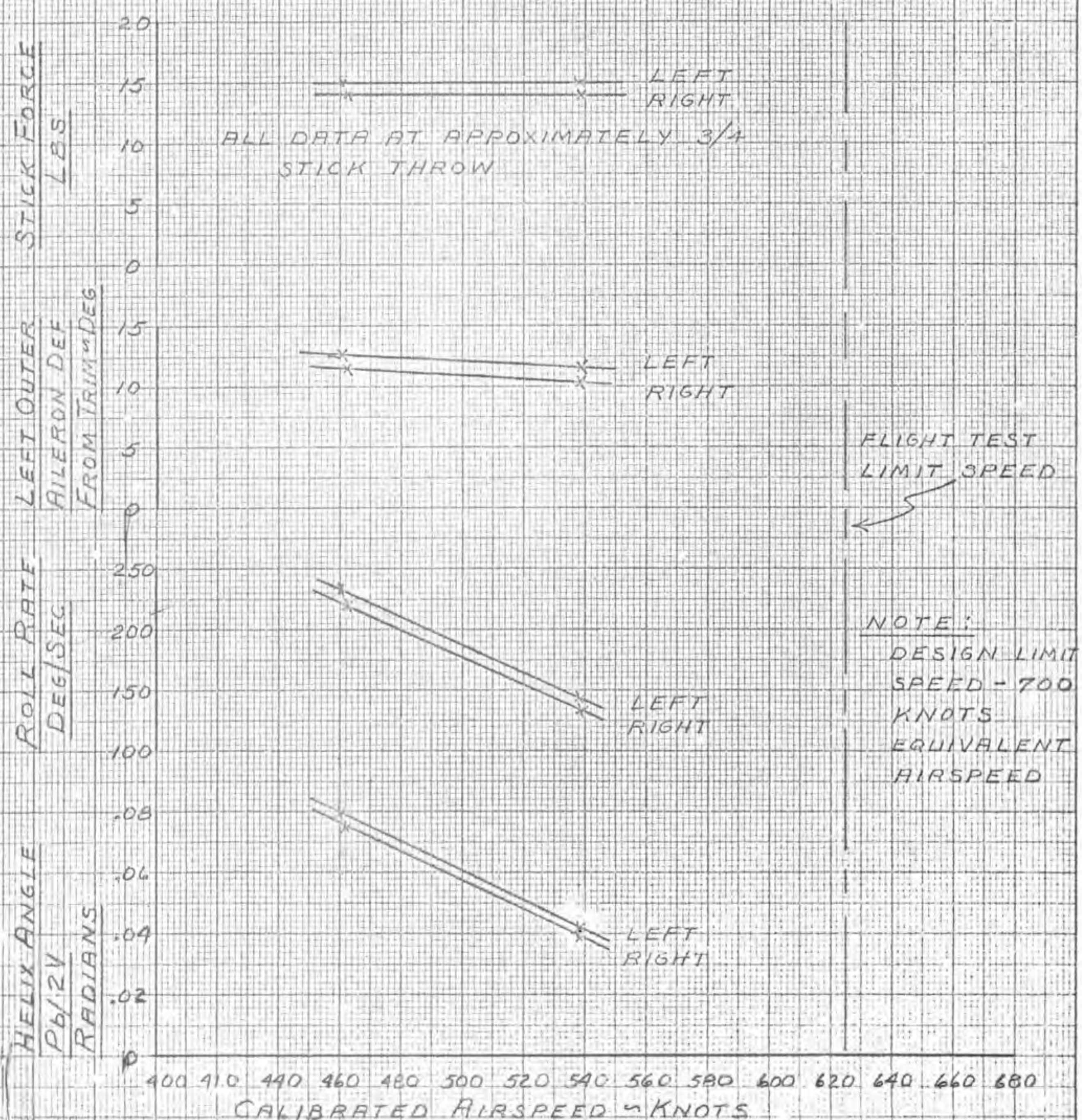


FIG NO 89
SUMMARY OF ROLLING CHARACTERISTICS
YF-100A USAF No 52-5754
POWER CONFIGURATION
POWER FOR LEVEL FLIGHT
RUDDER PEDALS FIXED
ALTITUDE ~ 11500 FEET
NO EXTERNAL STORES ATTACHED



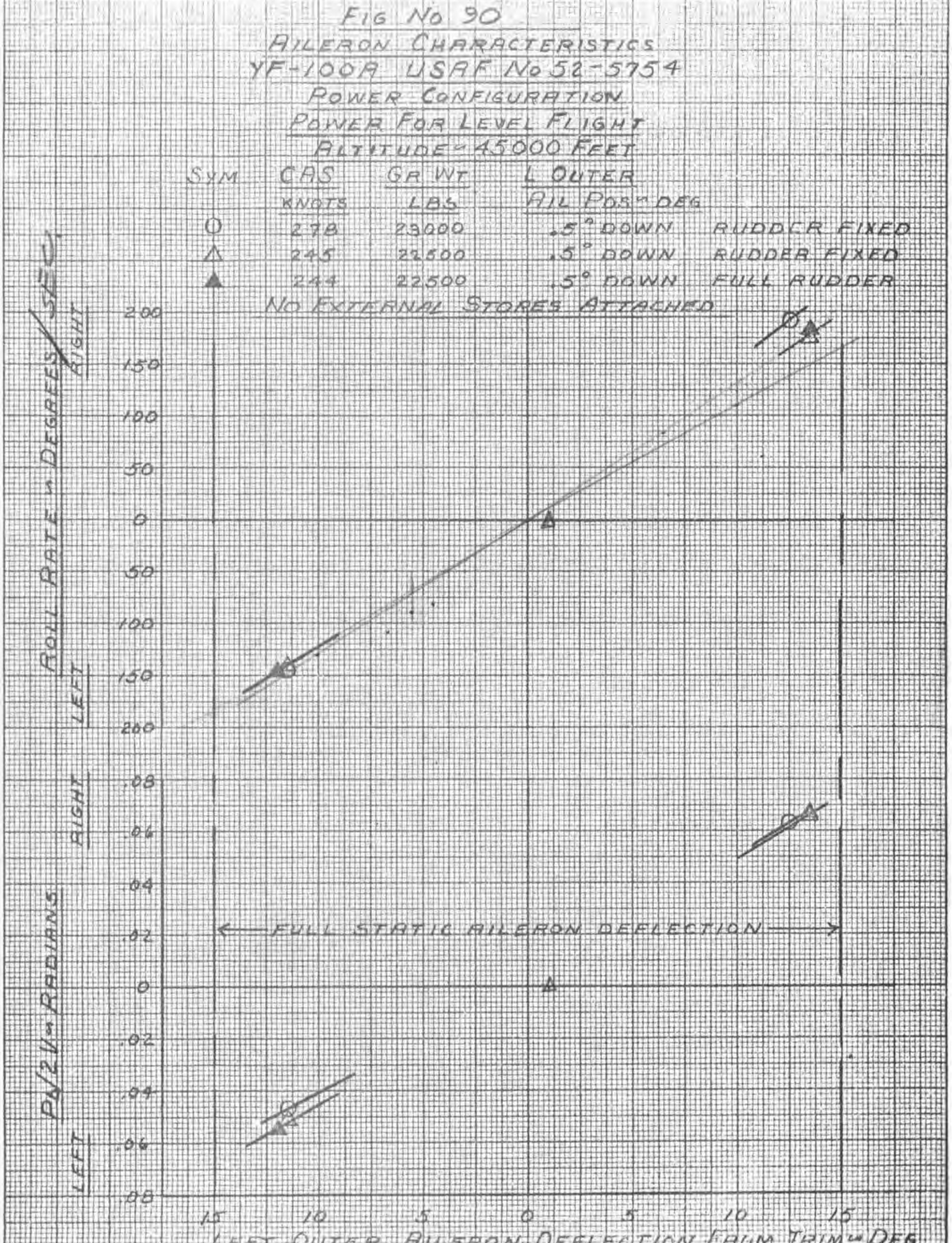


FIG NO 90
AILERON CHARACTERISTICS
YF-100A USAF No 52-5754
POWER CONFIGURATION
POWER FOR LEVEL FLIGHT
ALTITUDE ~ 45000 FEET
SYM CAS KNOTS GR WT LBS L OUTER AIL POS ~ DEG
○ 278 23000 .5° DOWN RUDDER FIXED
△ 245 22500 .5° DOWN RUDDER FIXED
▲ 244 22500 .5° DOWN FULL RUDDER
NO EXTERNAL STORES ATTACHED
ROLL RATE ~ DEGREES/SEC
RIGHT
LEFT
200
150
100
50
0
50
100
150
200
Pb/2V ~ RADIANS
.08
.06
.04
.02
0
.02
.04
.06
.08
FULL STATIC AILERON DEFLECTION
15
10
5
0
5
10
15
LEFT OUTER AILERON DEFLECTION FROM TRIM ~ DEG
UP
LEFT ROLL
DOWN
RIGHT ROLL

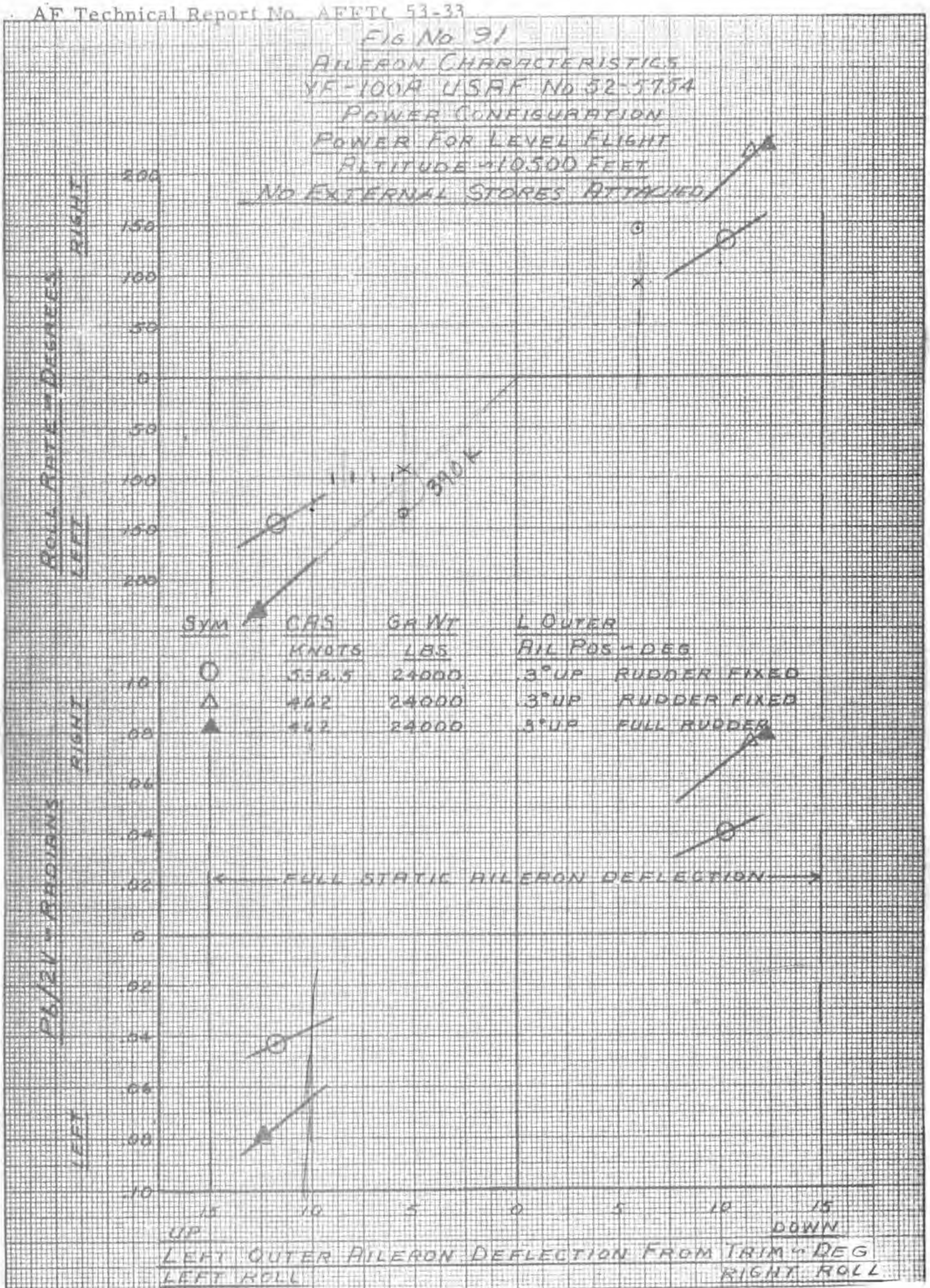


FIG NO 91
AILERON CHARACTERISTICS
YF-100A USAF NO 52-5754
POWER CONFIGURATION
POWER FOR LEVEL FLIGHT
ALTITUDE ~10500 FEET
NO EXTERNAL STORES ATTACHED
ROLL RATE ~ DEGREES
RIGHT
LEFT
200
150
100
50
0
50
100
150
200
390K
SYM
CAS KNOTS
GR WT LBS
L OUTER AIL POS ~ DEG
538.5
24000
.3° UP
RUDDER FIXED
442
24000
.3° UP
RUDDER FIXED
442
24000
.3° UP
FULL RUDDER
PB/2V ~ RADIANS
RIGHT
LEFT
.10
.08
.06
.04
.02
0
.02
.04
.06
.08
.10
FULL STATIC AILERON DEFLECTION
15
10
5
0
5
10
15
UP
DOWN
LEFT OUTER AILERON DEFLECTION FROM TRIM ~ DEG
LEFT ROLL
RIGHT ROLL

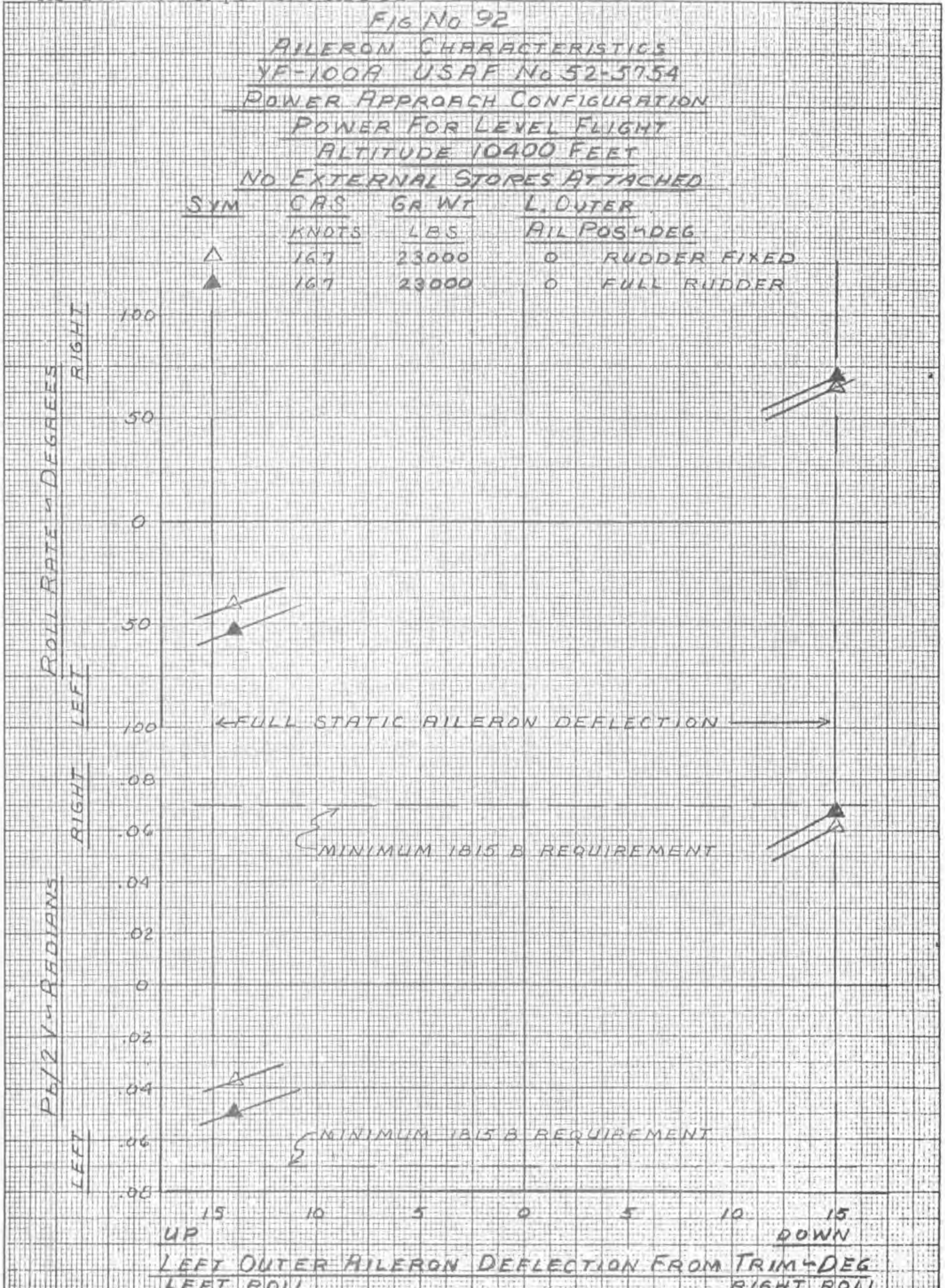


FIG NO 92
AILERON CHARACTERISTICS
YF-100A USAF No 52-5754
POWER APPROACH CONFIGURATION
POWER FOR LEVEL FLIGHT
ALTITUDE 10400 FEET
NO EXTERNAL STORES ATTACHED
SYM
CAS KNOTS
GR WT LBS
L. OUTER AIL POS ~ DEG
△ 167 23000 0 RUDDER FIXED
▲ 167 23000 0 FULL RUDDER
ROLL RATE ~ DEGREES
RIGHT
LEFT
100
50
0
50
100
←FULL STATIC AILERON DEFLECTION→
Pb/2V ~ RADIANS
.08
.06
.04
.02
0
.02
.04
.06
.08
MINIMUM 1815 B REQUIREMENT
MINIMUM 1815 B REQUIREMENT
15
10
5
0
5
10
15
UP
DOWN
LEFT OUTER AILERON DEFLECTION FROM TRIM ~ DEG
LEFT ROLL
RIGHT ROLL

Figure No. 93

MAXIMUM ALLOWABLE DEFLECTION

AILERON ROLLS

YF-100A, USAF No. 52-5754

Power Configuration(A/B On)

TRIM CONDITIONS

CAS 278 knots — Altitude 45,000 feet

CG 31.4 °/oMAC — Weight 23,000 lbs

Rudder Pos. 1.5° left — L. Outer Ail. Pos. .5° Dn

No external stores attached

LEFT ROLL — RIGHT ROLL

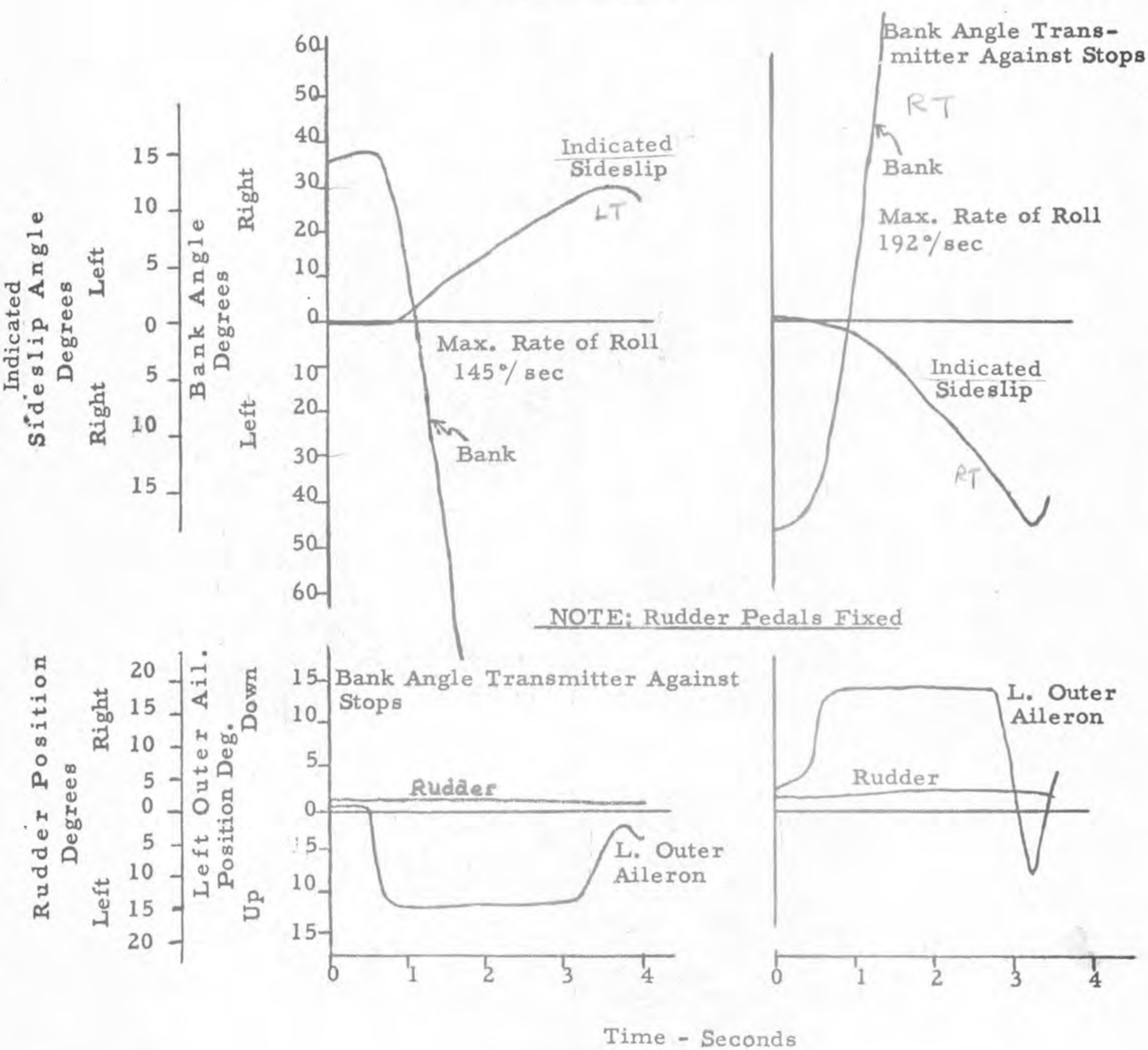


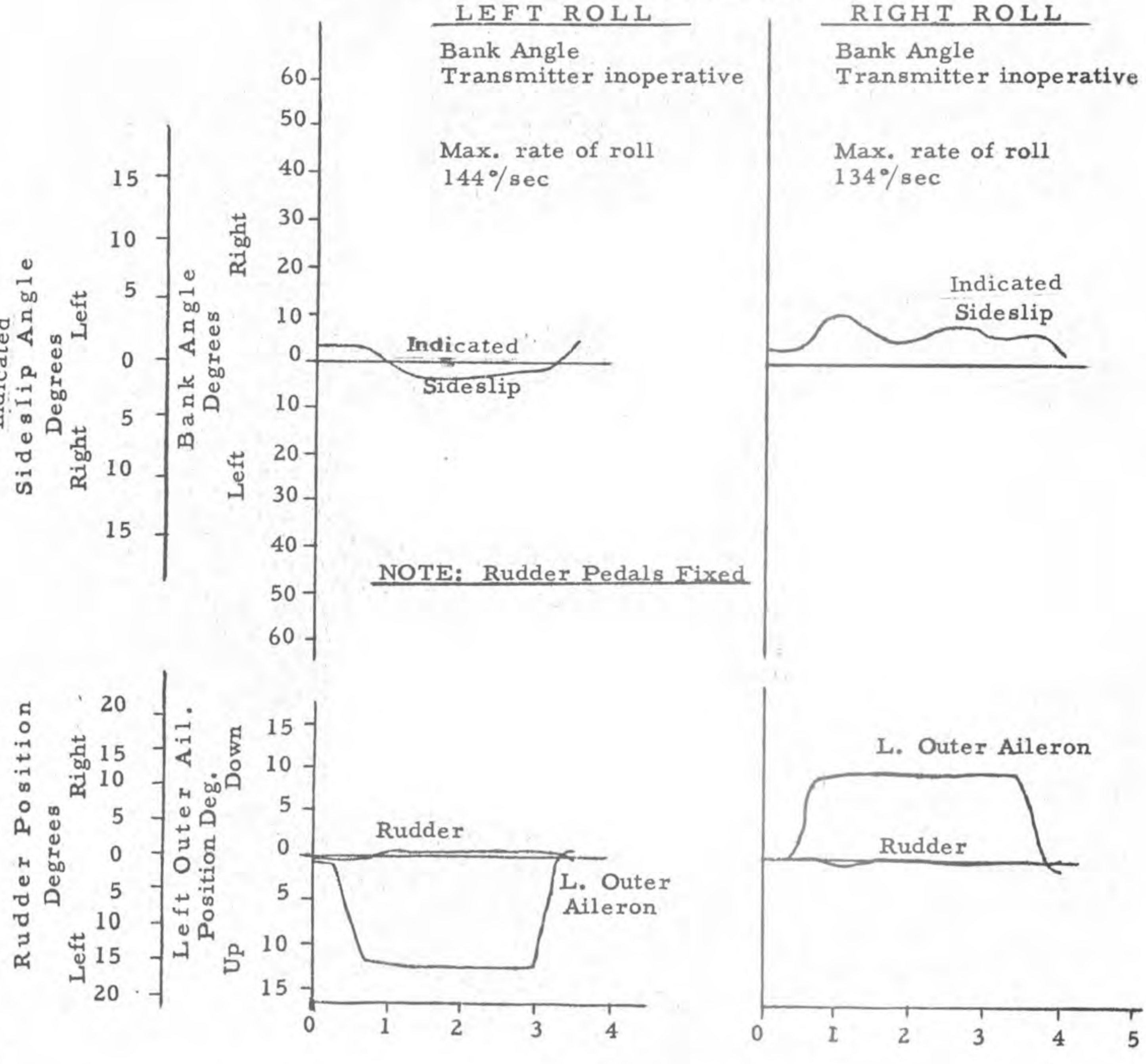
Figure No. 94

MAXIMUM ALLOWABLE DEFLECTION
AILERON ROLLS

YF-100A, USAF No. 52-5754
Power Configuration(A/B On)

TRIM CONDITIONS

CAS 538.5 knots
CG 30.4 °/o MAC
Rudder Pos. 0
Altitude 10,500 feet
Weight 24,000 lbs
L. Outer Ail. Pos. .3° up
No external stores attached



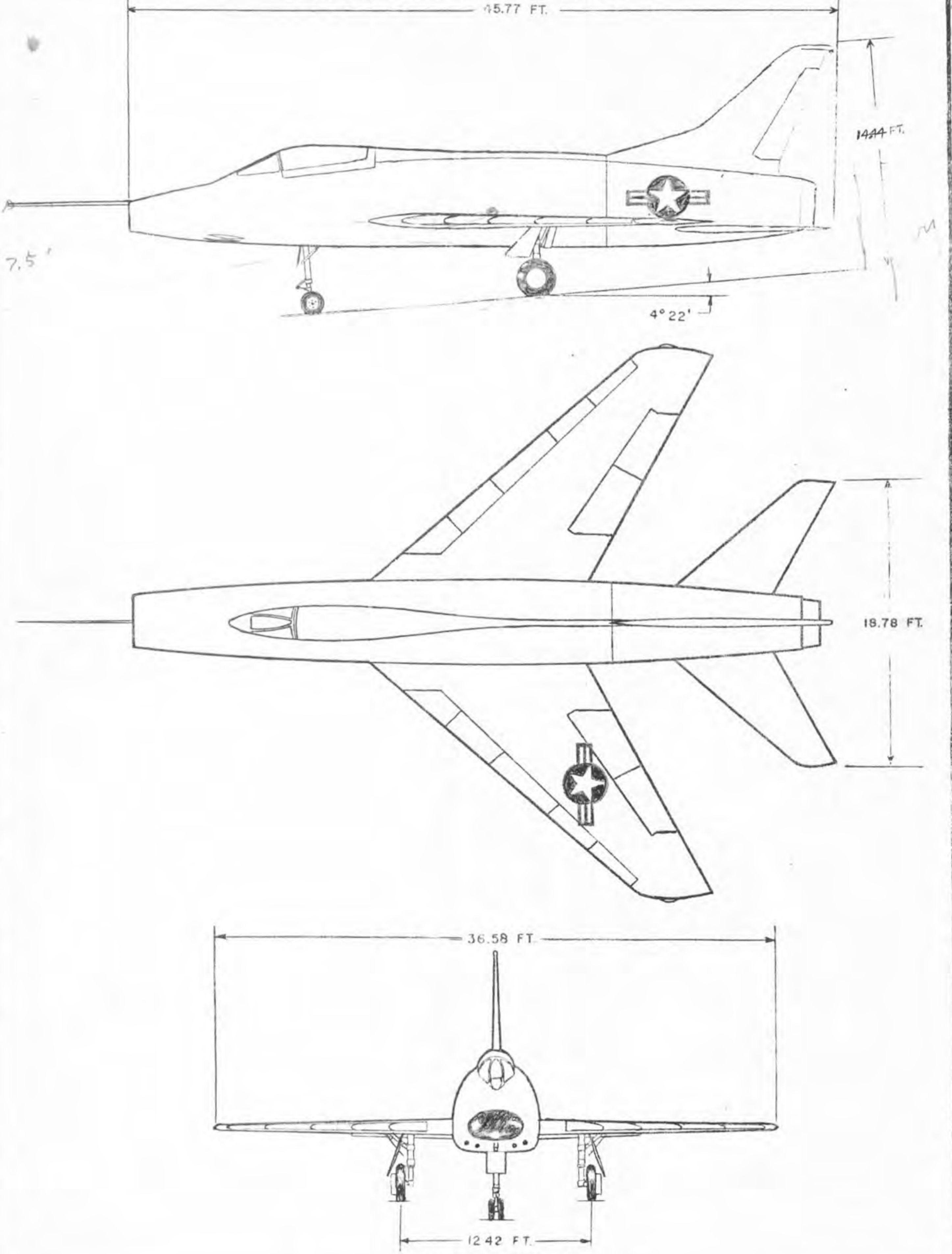
Time - Seconds

APPENDIX II

INDEX

General Information and Photographs
YF100A USAF, No. 52-5754

THREE VIEW DRAWING



APPENDIX II

B. DIMENSIONS AND DESIGN DATA

The following dimensions and design data were obtained from North American Aviation Inc., Report No's NA-53-439 and NA-51-857.

1. Wing

Area	376.02 ft.
Span	36.58 ft.
Sweepback of 25 per cent element	45°
Aspect ratio	3.56
Taper ratio	.30
Mean aerodynamic chord	135.93
Fuselage station of 25 per cent $\overline{C}_w$(WP -23.00)	301.97
Dihedral angle	0°
Incidence angle of root chord	0°
Incidence angle of tip chord	0°
Airfoil sections, root and tip, NACA 64A007 (parallel to plane of symmetry)	

2. Ailerons

(Flat sided type)	
Area (aft of hinge line), each	19.32 sq. ft.
Area moment (normal to hinge line)	41.51 cu. ft.
Span	93.67 in.
Spanwise location, inboard end	.323 $b_w/2$
Spanwise location, inboard end	.750 $b_w/2$
Ratio of aileron chord to wing chord	.250
Deflection, maximum	15° up, 15° down
Irreversible hydraulic boost and artificial feel	
Aerodynamic balance	none
Static balance	internal, lead weights

3. Leading Edge Slats

Span, equivalent	152.52 in.
Segments	5
Spanwise location, inboard end	.246 $b_w/2$
Spanwise location, outboard end	.941 $b_w/2$
Ratio of slat chord to wing chord (Parallel to F.R.L.)	.200
Rotation, maximum	10°
Center of rotation(measured in plane containing 13 per cent element line, downward from and normal to W.R.P.)	263 per cent slat chord

4. Horizontal Tail

Area	99.00 sq. ft.
Span	18.78 ft.
Sweepback of 25 percent element	45°
Aspect ratio	3.56
Taper ratio	.30
Mean aerodynamic chord	69.72 in.
Fuselage station of $.25\bar{C}_w$(WP -28.97)	479.64
Dihedral angle	0°
Airfoil sections, root and tip, NACA 64A007(parallel to plane of symmetry)	

Check this for F-100D A/C

5. Horizontal stabilizer

Area	67.35 sq. ft.
Area moment (normal to C_L rotation)	337.36 cu. ft.
Deflection, maximum	5° nose up, 25° nose down
Axis of rotation	W.L. -22 F.S. 485

Irreversible hydraulic boost and artificial feel

6. Vertical Tail

Area (excluding dorsal fin)	39.84 sq. ft.
Span, unblanketed	7.46 ft.
Sweep of 25 percent element	45°
Aspect ratio	1.49
Taper ratio	.30
Mean aerodynamic chord	66.08 in.
Fuselage station of $25\bar{C}_V$(WP 57.63)	488.70
Airfoil sections, root and tip, NACA 64A007(parallel to F.R.L.)	

7. Vertical Fin

Area(excluding dorsal fin)	33.54 sq. ft.
Area of dorsal fin	3.16 sq. ft.
Span, unblanketed	7.46 ft.
Offset	0°

8. Rudder

Area (aft of hinge line)	6.30 sq. ft.
Area moment (normal to hinge line)	6.37 cu. ft.

Span	5.77 ft.
Spanwise location, inboard end	.031 b_v
Spanwise location, outboard end	.804 b_v
Ratio of rudder chord to vertical tail chord	.200
Deflection, maximum	20° L, 20° R
Aerodynamic balance	Overhanging, unsealed

9. Fuselage

Side area (total)	277.86 sq. ft.
Length(afterburner nozzle closed)	45.03
Depth over canopy, maximum	76.50
Width, maximum	67.00
Fineness ratio(afterburner nozzle closed)	7.75

10. Speed Brake

Area, surface	14.14 sq. ft.
Deflection, maximum	50°
Fuselage station of hinge line (WP -41.00)	223.75

11. Total Surface Area of Clean Airplane 1519.60 sq. ft.

12. Total Frontal Area of Clean Airplane 57.34 sq. ft.

13. Landing Gear

Main gear tire size	30 x 8.8
Nose gear tire size	dual 18 x 4.4
Hydraulic steering	
Nose gear steering angle	35°L, 35°R

14. External Stores

275 gallon drop tanks	2
Length	250 in.
Diameter, maximum	25 in.
Fin incidence	2°

C. OPERATIONAL LIMITATIONS

1.	Structural design gross weight at combat radius (clean aircraft)		23,996 lbs.
2.	Design load factors at (1 above) for symmetrical flight are:	Positive	+7.33
		Negative	-3.33
	Note: because structural integrity tests have not yet been made on the YF-100A aircraft, limit load factors for Phase II tests were given by the contractor as:	Positive	+5
		Negative	-2
3.	Maximum structural design flight weight (external tanks)		28,561 lbs.
4.	Design load factors at (3 above) for symmetrical flight are:	Positive	+6
		Negative	-3
	Flight test load factors:	Positive	+4
		Negative	-2
5.	Maximum take-off weight		28,561 lbs.
6.	Minimum flight weight		19,448 lbs.
7.	Ground load factors: Design landing weight of		22,205 lbs.
	Main gear 3.0		
	Nose gear 3.0		
8.	Limit airspeeds		
	(a) Clean aircraft above 3,000 feet		700 knots
	(b) With external fuel tanks above 3,000 feet		600 knots
	Note: limit speed allowed for flight test was		625 knots IAS

9. Most forward aft cg position possible in flight:

<u>Clean</u>

Fwd.	29.5 per cent of MAC
Aft	35.5 per cent of MAC

D. FUEL SYSTEM

1. Main system (internal). Fuel is supplied to the engine from five bladder

cells interconnected to form three tanks with capacities as follows:

a.	sump tank (forward cell and underwing cell)	417 gal.
b.	intermediate tank (left and right intermediate cells)	221 gal.
c.	aft tank	119 gal.
	total capacity	757 gal.

2. External fuel system:

Provisions are made for the installation of a 275 gallon droppable combat fuel tank beneath each wing.

total capacity 2 tanks 550 gal.

E. POWER PLANT

PRATT and WHITNEY XJ57-P7 ENGINE

Power Conditions	Thrust lbs	SFC
Uninstalled maximum rating with afterburning	13,200	1.98
Uninstalled military rating without afterburning	8,450	.86
Installed maximum rating with afterburning (9,495 rpm)*	10,200	2.2
Installed military rating without afterburning (9,495 rpm)*	6,800	.91

* rpm values are based on a compressor inlet temperature of 15°C.

F. WEIGHT AND BALANCE

Flight no.	Basic* Weight-lbs	Crew lbs	Other lbs	Gross Wt lbs	CG ** % MAC
1-27	25,169	200	0	25,369	31.9
28	25,169	200	24[1]	25,393	32.0
29	23,062[2]	200	24[1]	23,286	31.8
30	22,490[2]	200	24[1]	22,714	31.1
31-33	25,169	200	24[1]	25,393	32.0
34	25,169	200	0	25,369	31.9
35	25,169	200	24[1]	25,393	32.0
36	25,169	200	0	25,369	31.9
37	25,169	200	400[3]	25,769	32.2
38-39	25,169	200	424[4]	25,793	32.4

* Includes test instrumentation, 1289 lbs ballast, 757 gallons of fuel and 5.5 gallons of oil.

**CG computed with gear up; add 0.1 percent MAC for gear down percent MAC.

(1.) Drag chute and deploy cable.

(2.) Partial fuel load.

(3.) Two empty 275 gallon external fuel tanks.

(4.) Two empty 275 gallon external fuel tanks and drag chute and deploy cable.

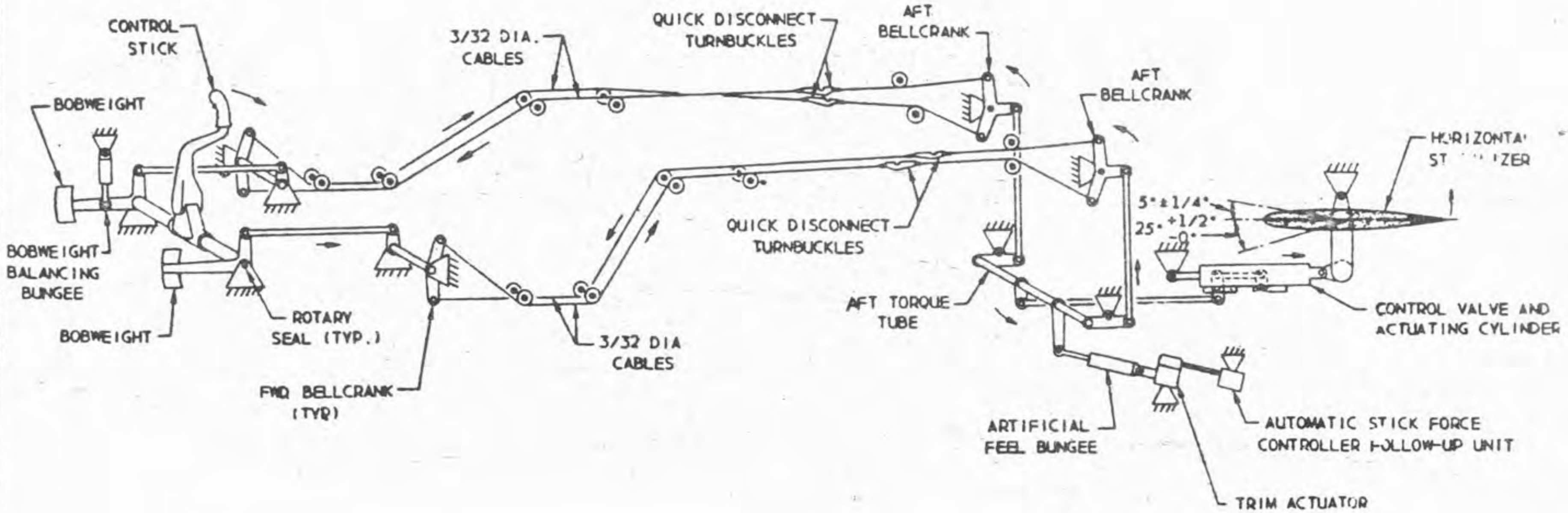
G. TEST INSTRUMENTATION

Complete instrumentation for performance and stability and control was installed in the test aircraft by the contractor. All instrument calibrations were accomplished by the contractor. The calibrations on several of the more critical instruments were checked by the AFFTC Instrumentation Branch. The following instruments were used for the Phase II tests:

1. Pilot's panel:
   - Airspeed
   - Altimeter
   - Free air temperature
   - Engine rpm (high pressure)
   - Exhaust gas temperature
   - Sideslip angle
   - Fuel quantity (forward cell)
   - Fuel quantity (intermediate cell)
   - Fuel quantity (aft cell)
   - Accelerometer
   - Engine fuel counter
   - A/B fuel counter
   - Tail pipe nozzle position

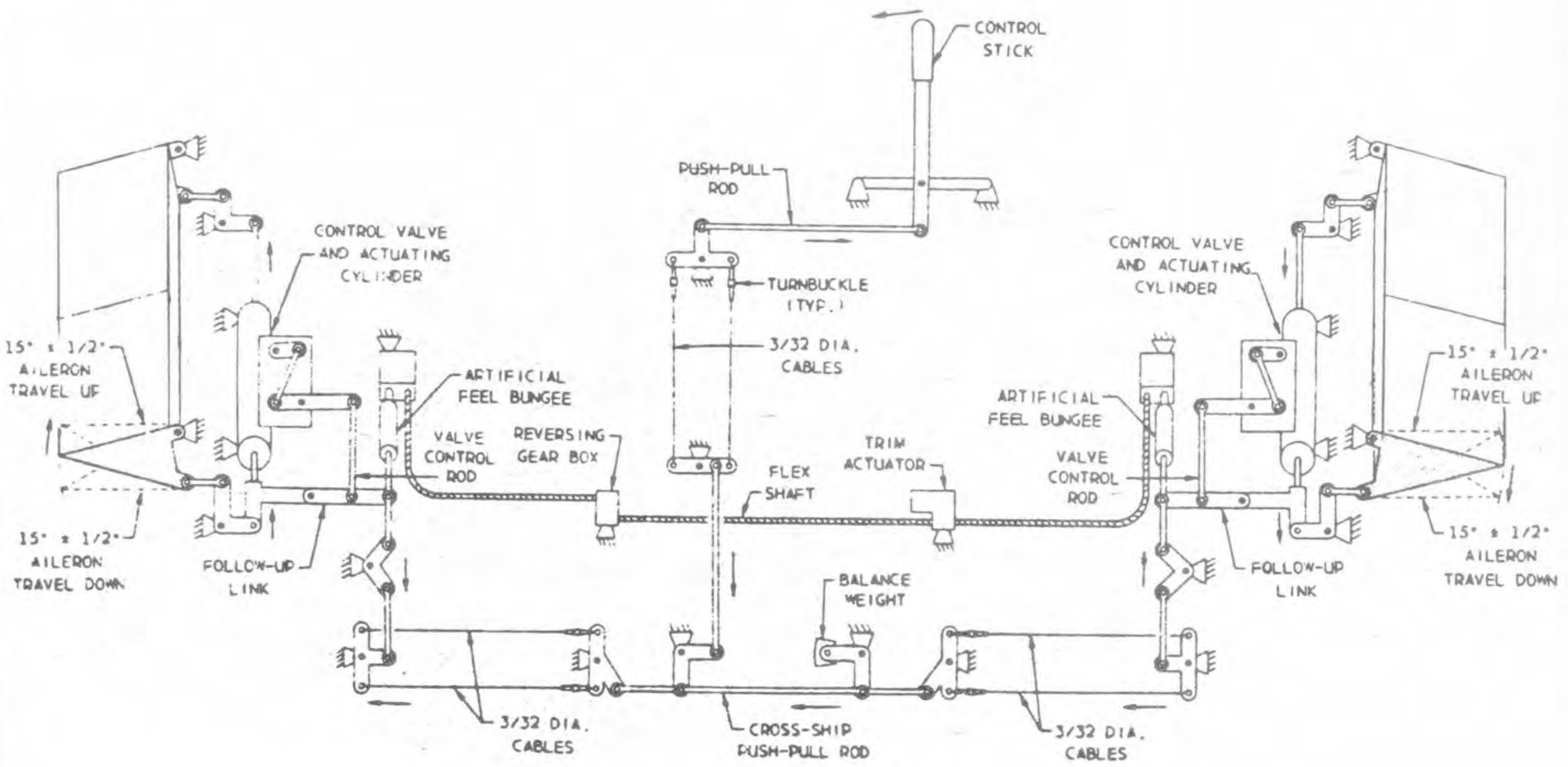
2. 35 mm Photorecorder:
   - Airspeed
   - Altitude
   - Free air temperature
   - Engine fuel counter
   - A/B fuel counter
   - Aft fuel cell quantity
   - Engine rpm (high pressure)
   - Engine rpm (low pressure)
   - Tailpipe temperature
   - Tailpipe total head pressure
   - Compressor inlet total head pressure
   - Fuel pressure at outlet of A/B control
   - Slat position
   - Vertical accelerometer
   - Clock
   - Oil cooler duct flap position
   - Nozzle position light
   - Surge bleed valve position lights

3. Eighteen channel Oscillograph:
   - Normal acceleration
   - Horizontal stabilizer position
   - Longitudinal stick position
   - Left aileron position (inboard)
   - Left aileron position (outboard)
   - Rudder position
   - Longitudinal stick force
   - Aileron stick force
   - Rudder pedal force
   - Speed brake position
   - Horizontal stabilizer trim actuator position
   - Angle of bank
   - Angle of attack
   - Sideslip angle
   - Photorecorder and pilot correlation trace
   - Nose wheel lift-off
   - Power control lever position



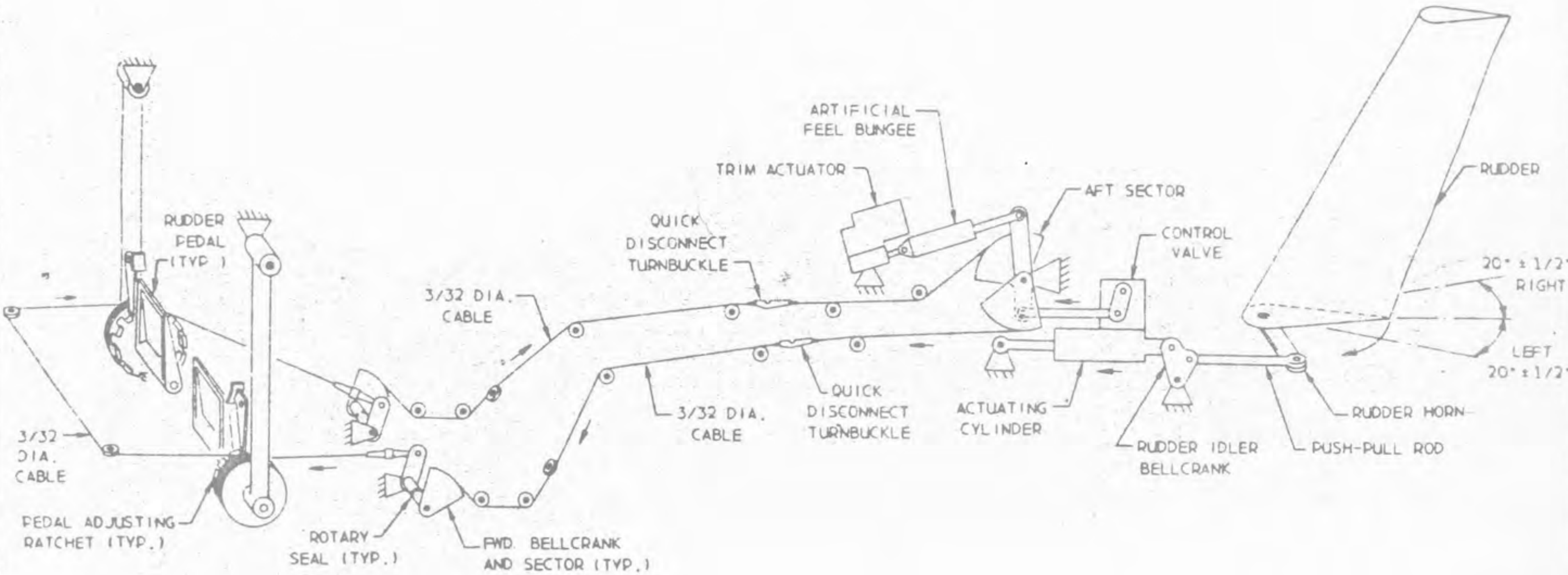
HORIZONTAL STABILIZER CONTROL SYSTEM

YF-100A



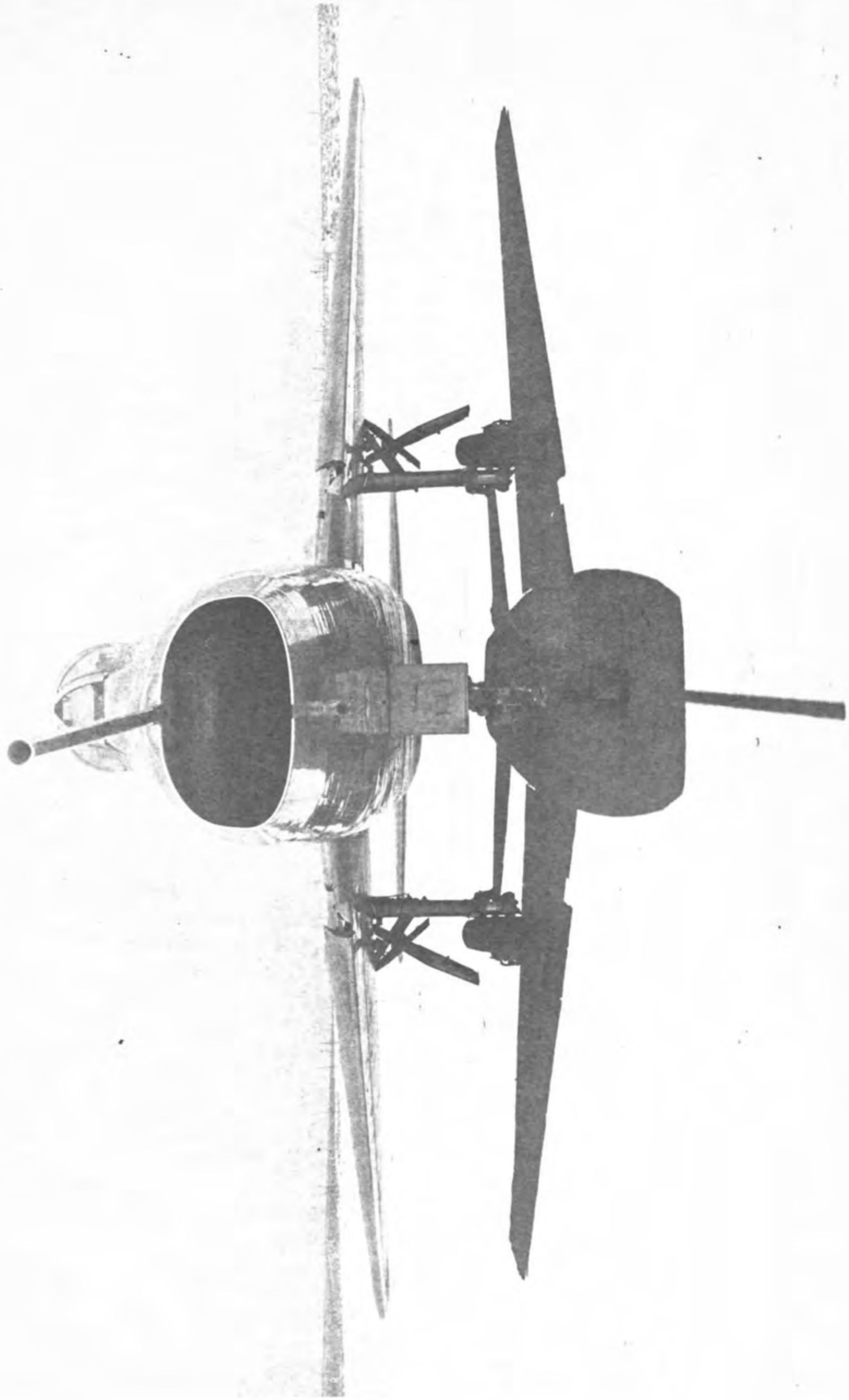
AILERON CONTROL SYSTEM

YF-100A

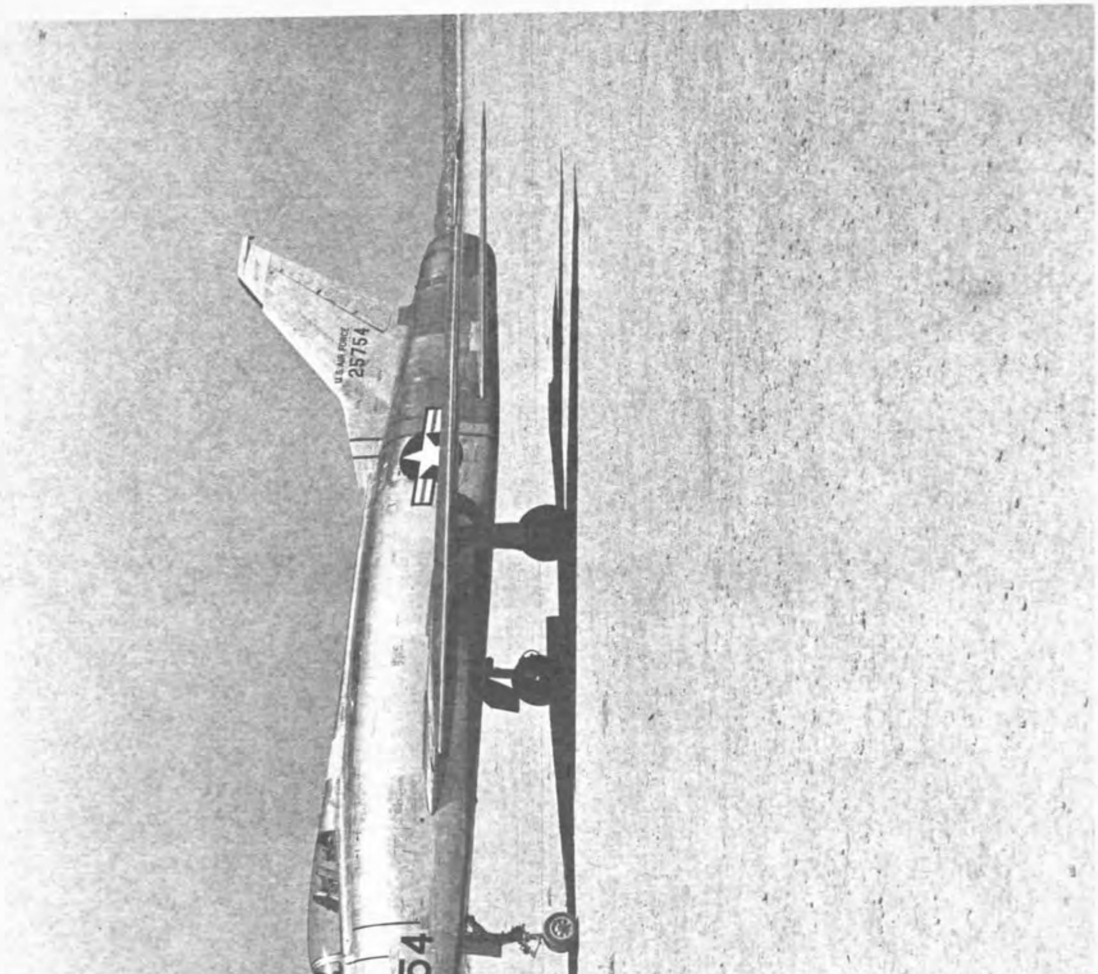


RUDDER CONTROL SYSTEM

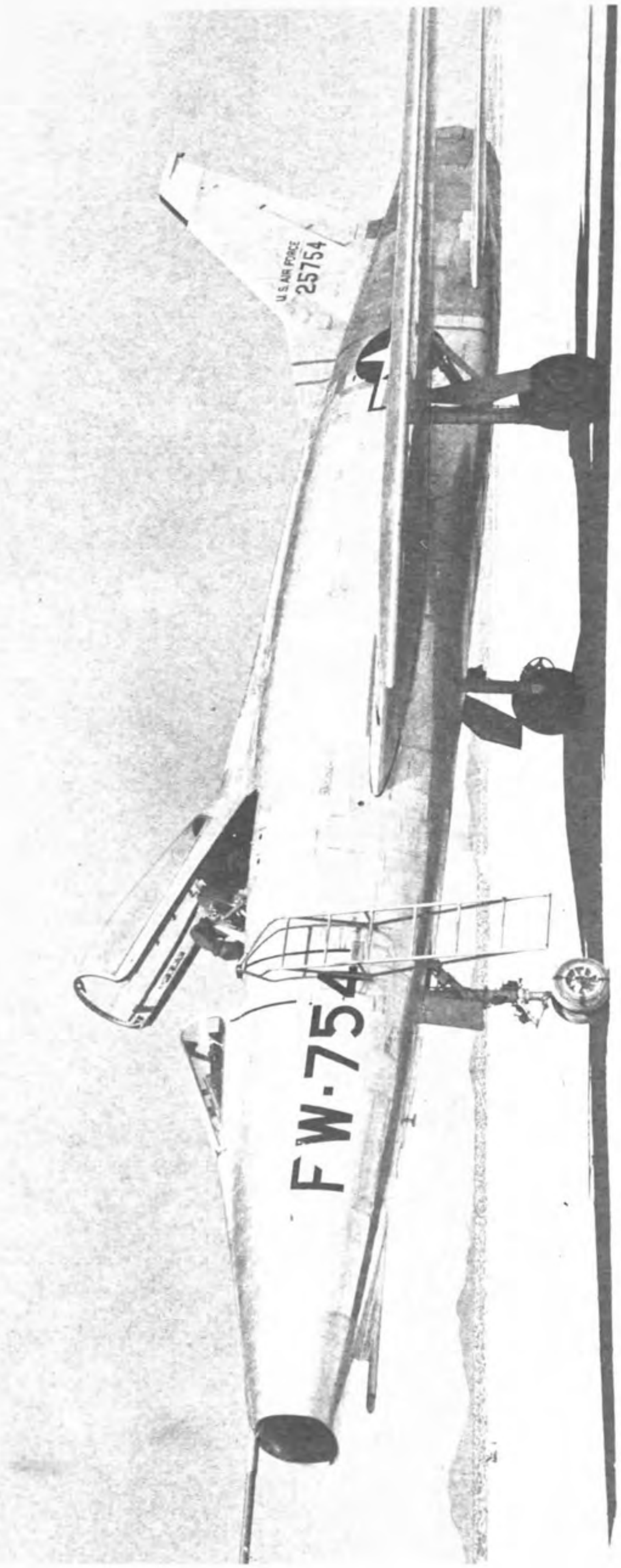
YF-100A



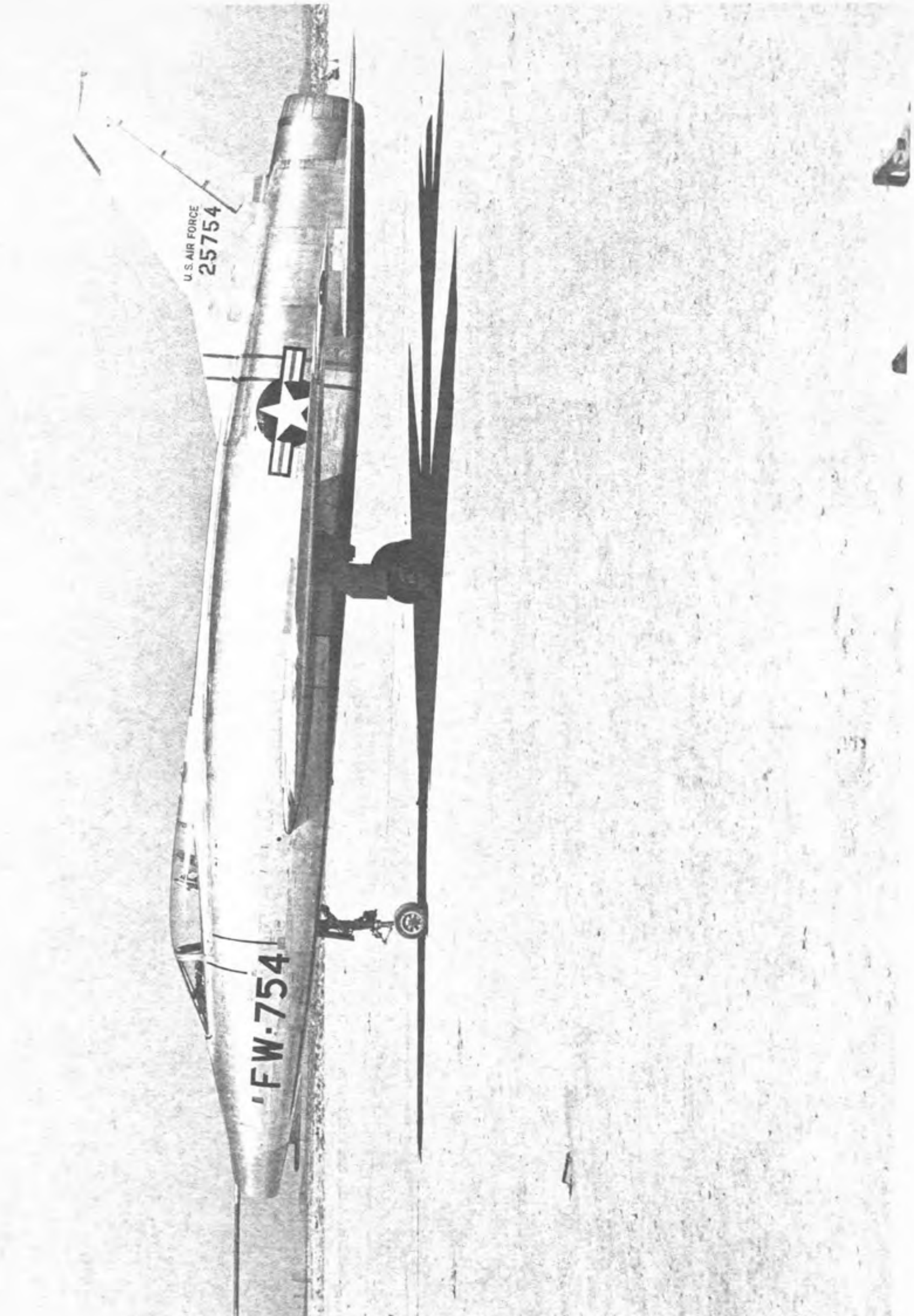
FRONT VIEW



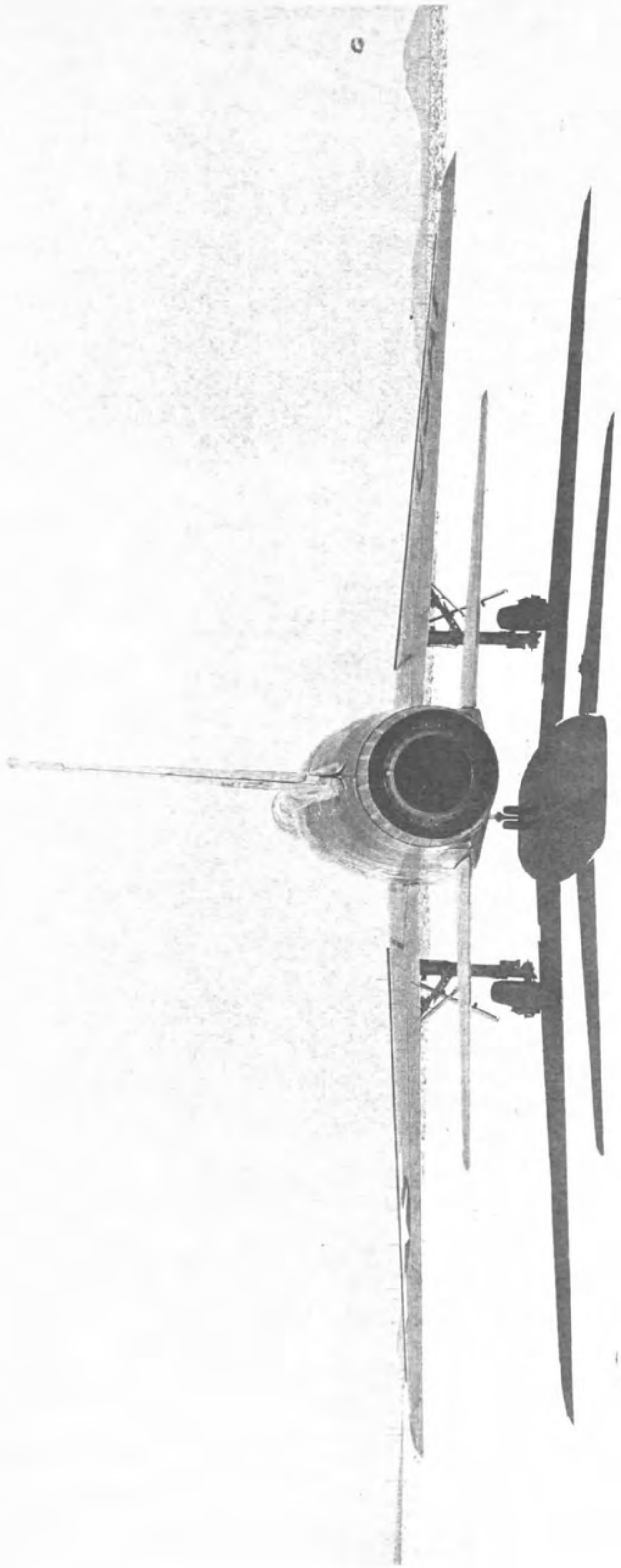
LEFT FRONT VIEW



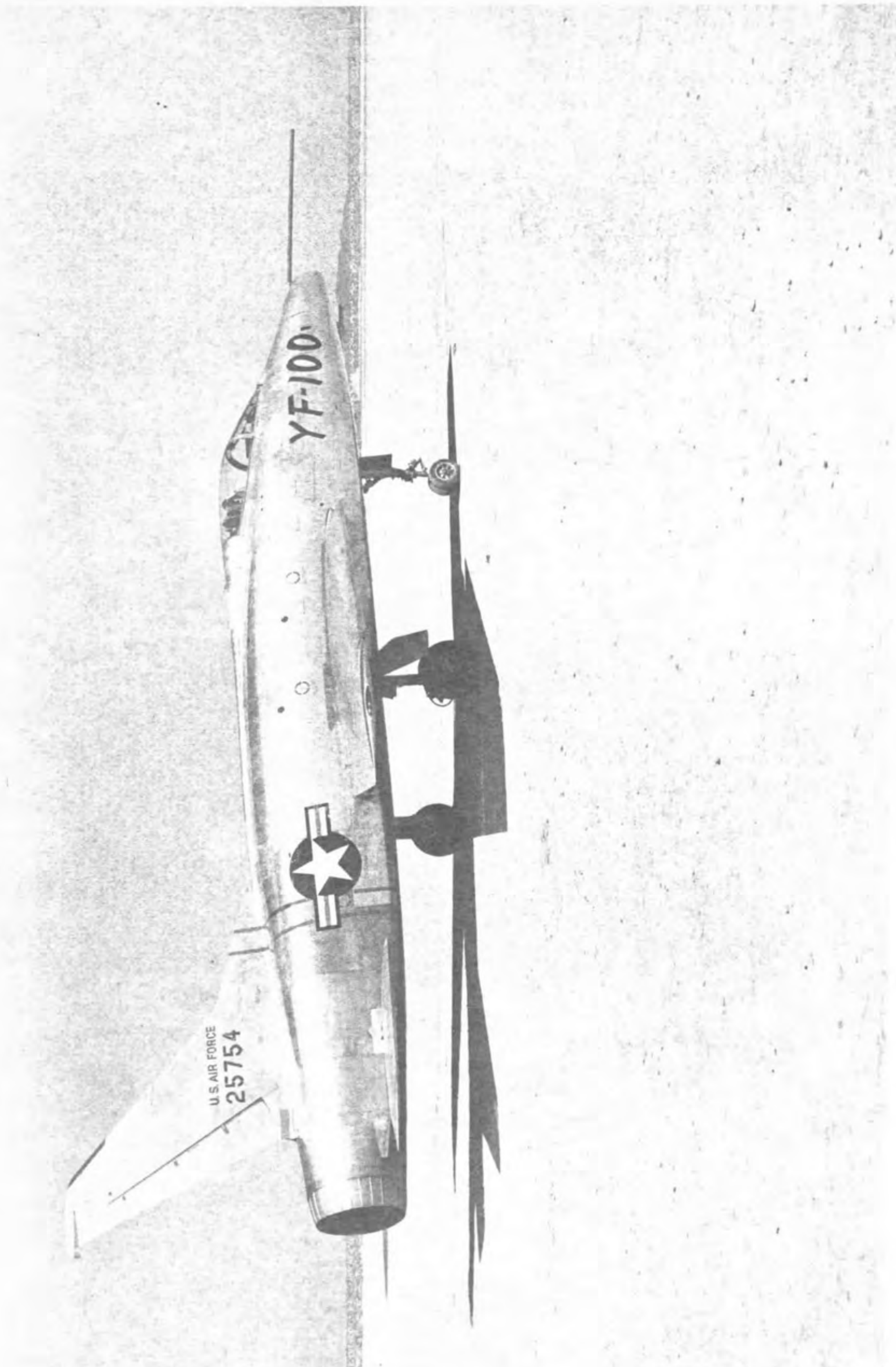
LEFT FRONT VIEW (Canopy open and ladder attached)



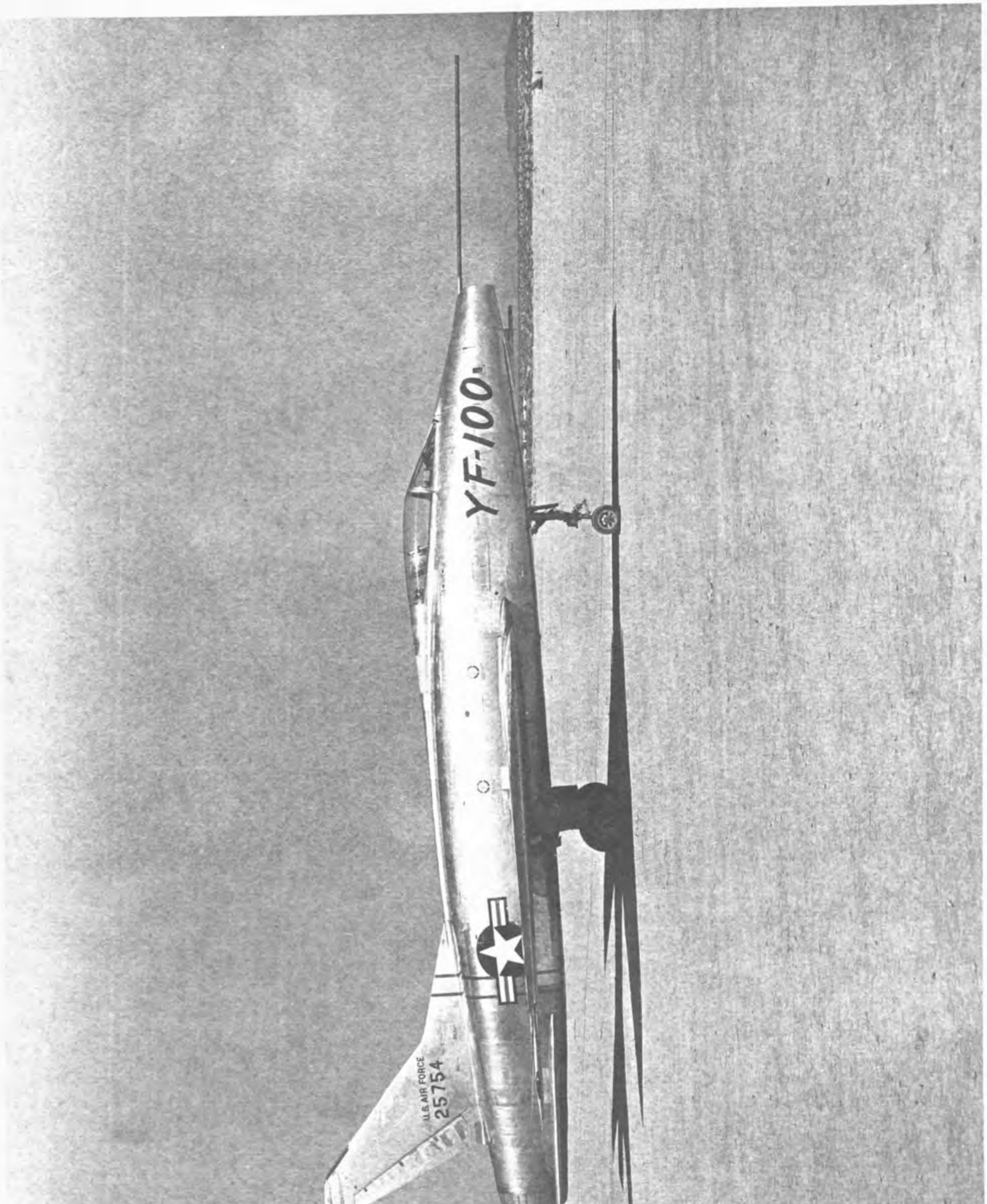
LEFT SIDE VIEW



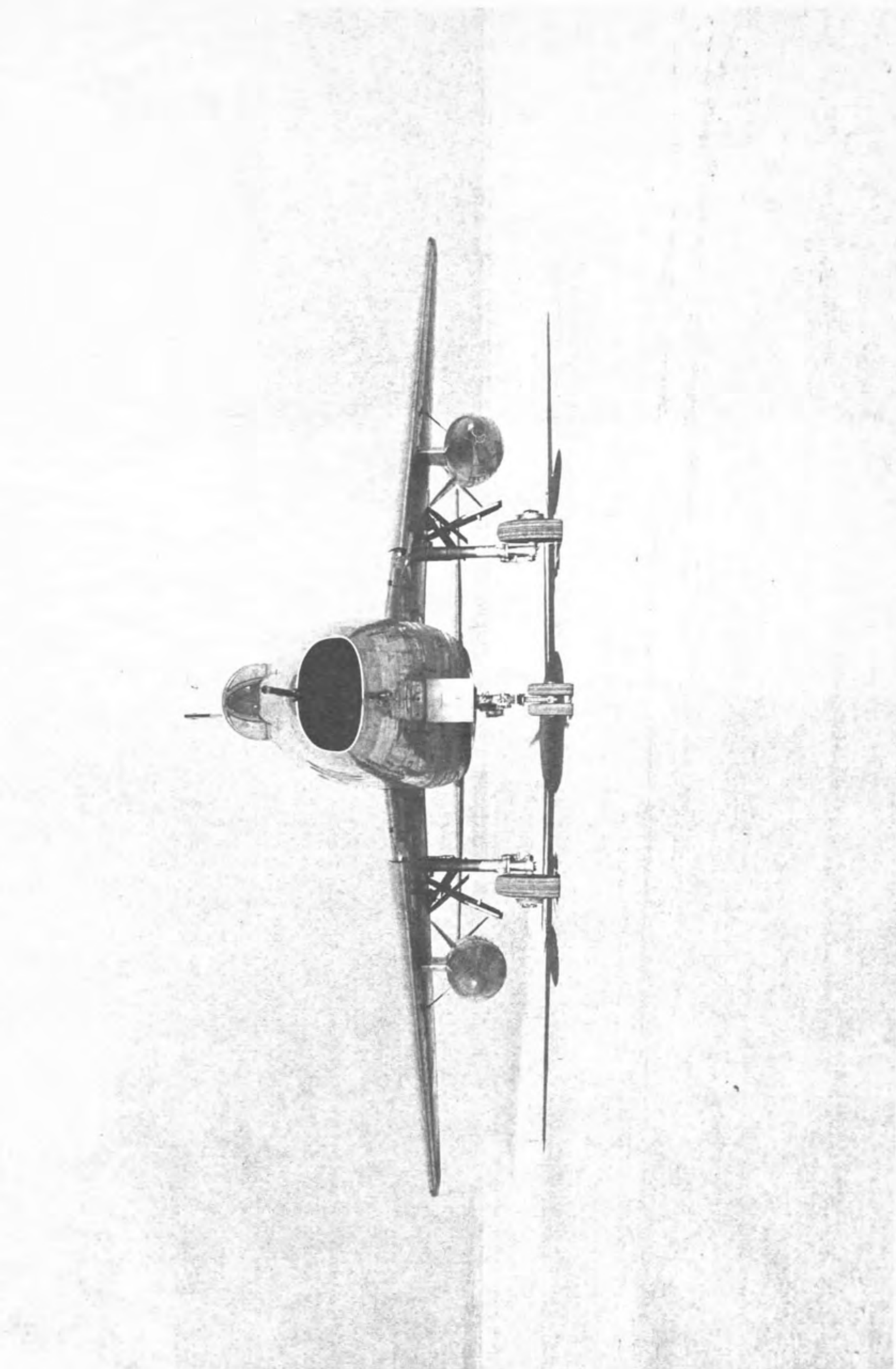
REAR VIEW



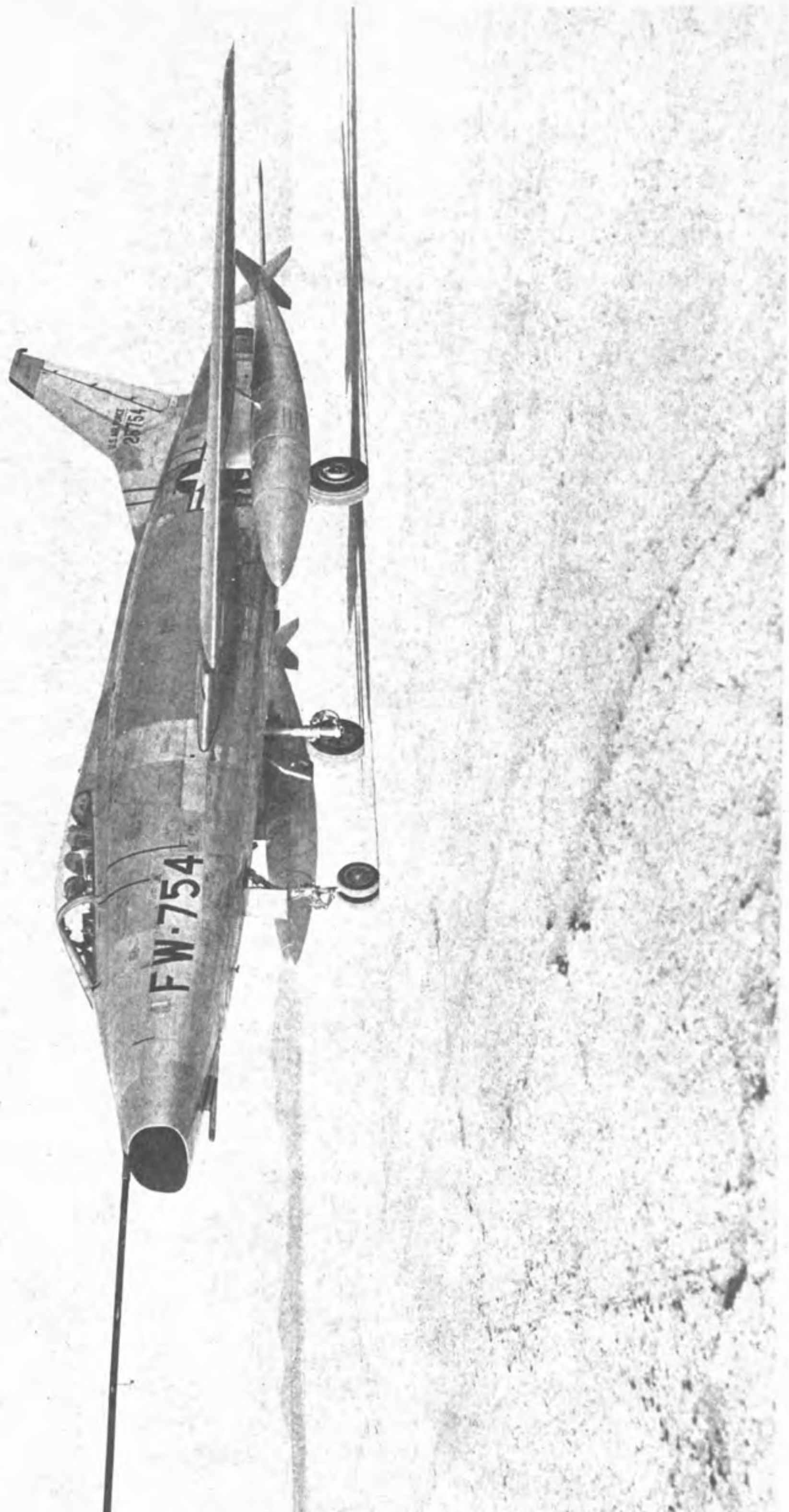
RIGHT REAR VIEW



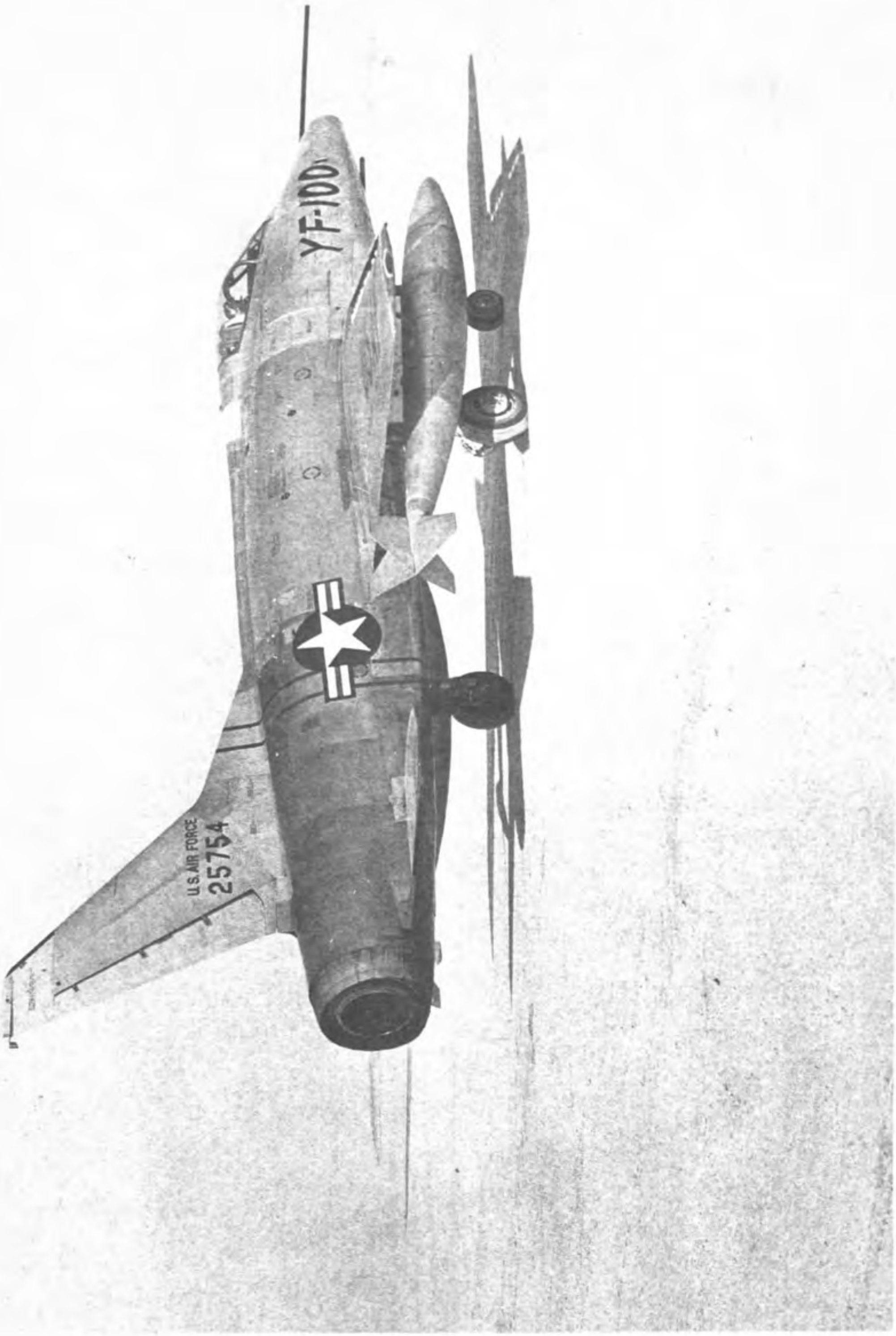
RIGHT SIDE VIEW



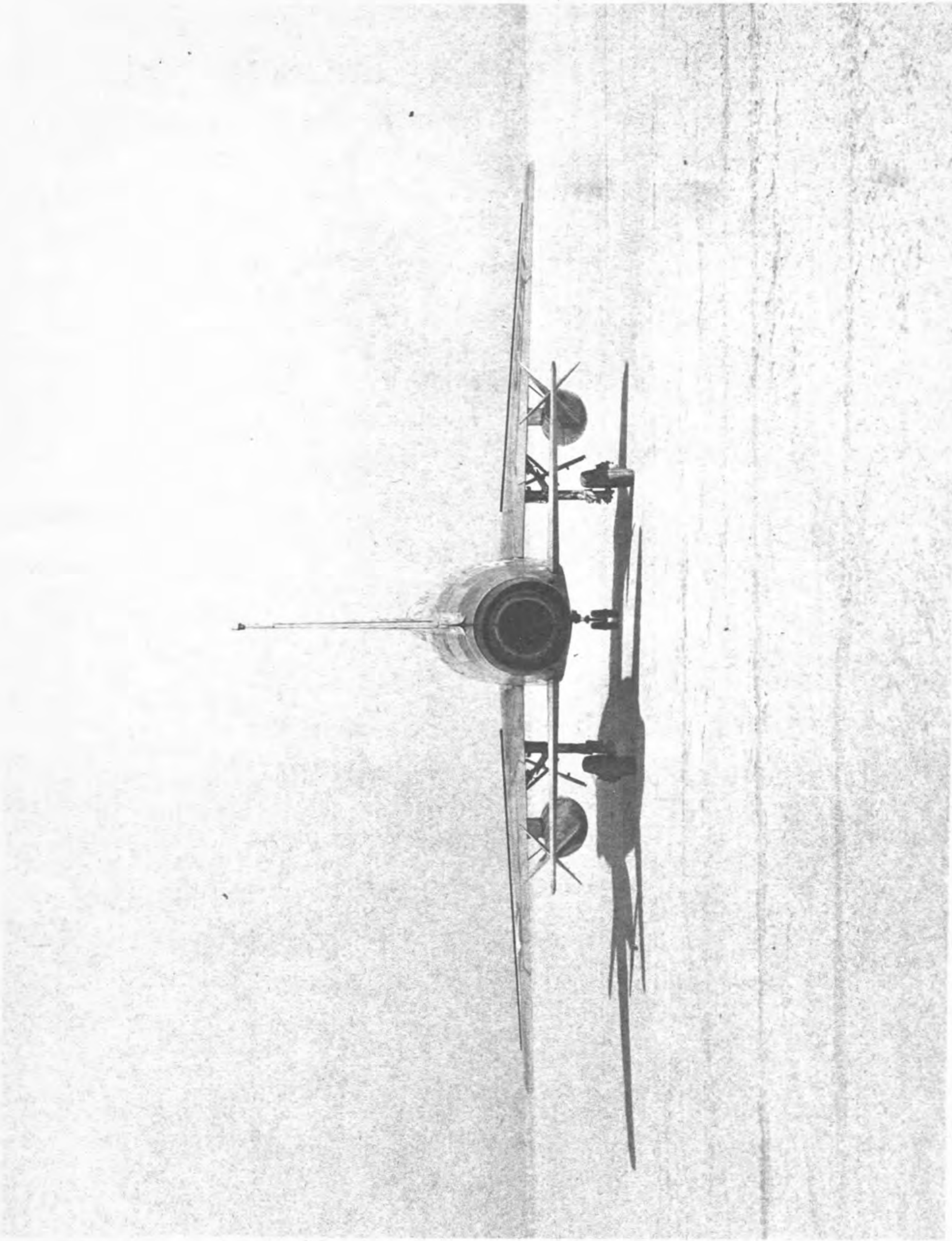
FRONT VIEW (Tanks on)



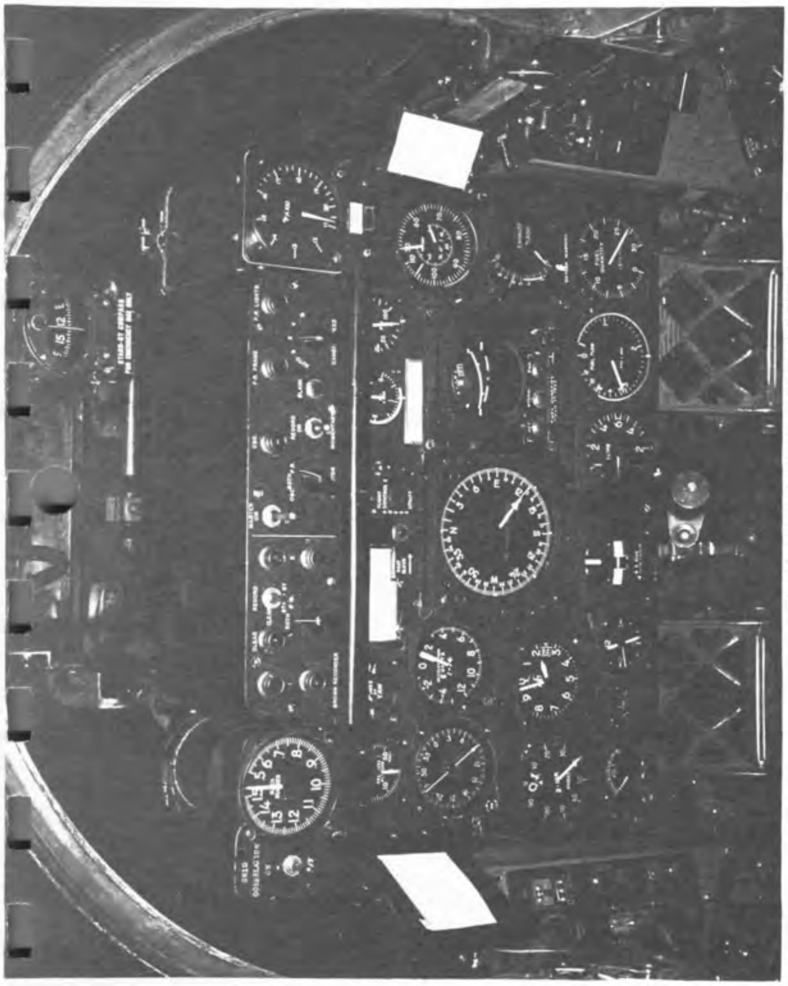
LEFT FRONT VIEW (Tanks on)



RIGHT REAR VIEW(Tanks on)



REAR VIEW (Tanks on)

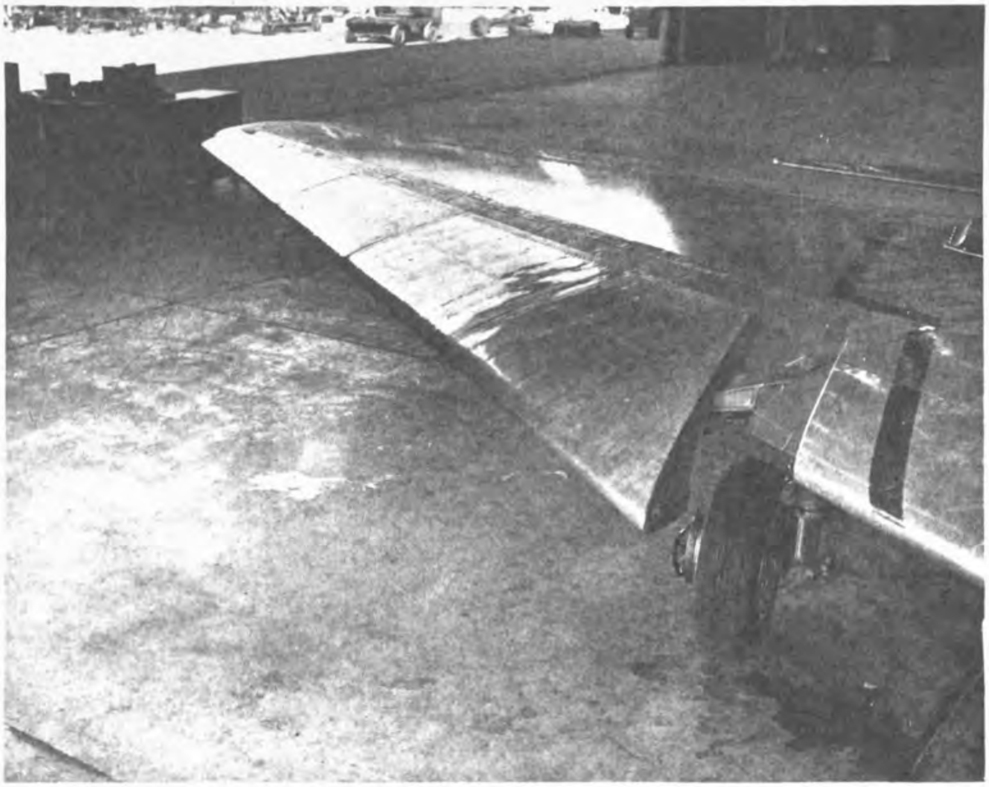


INSTRUMENT PANEL

RIGHT CONSOLE

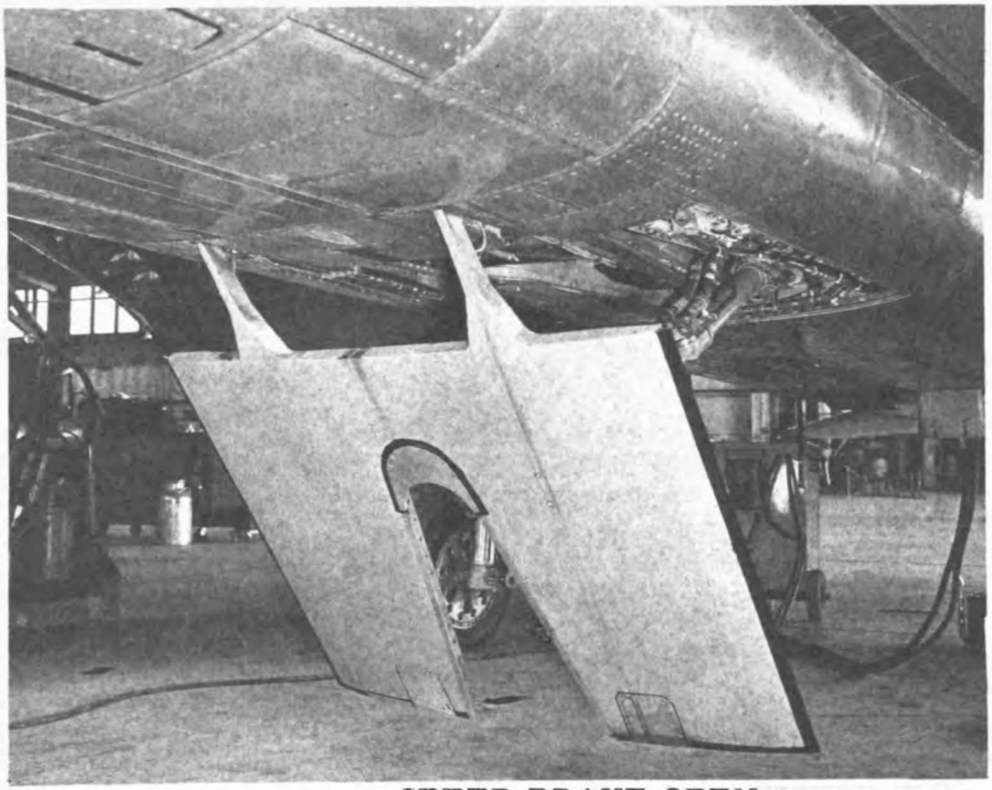
LEFT CONSOLE

COCKPIT LAYOUT

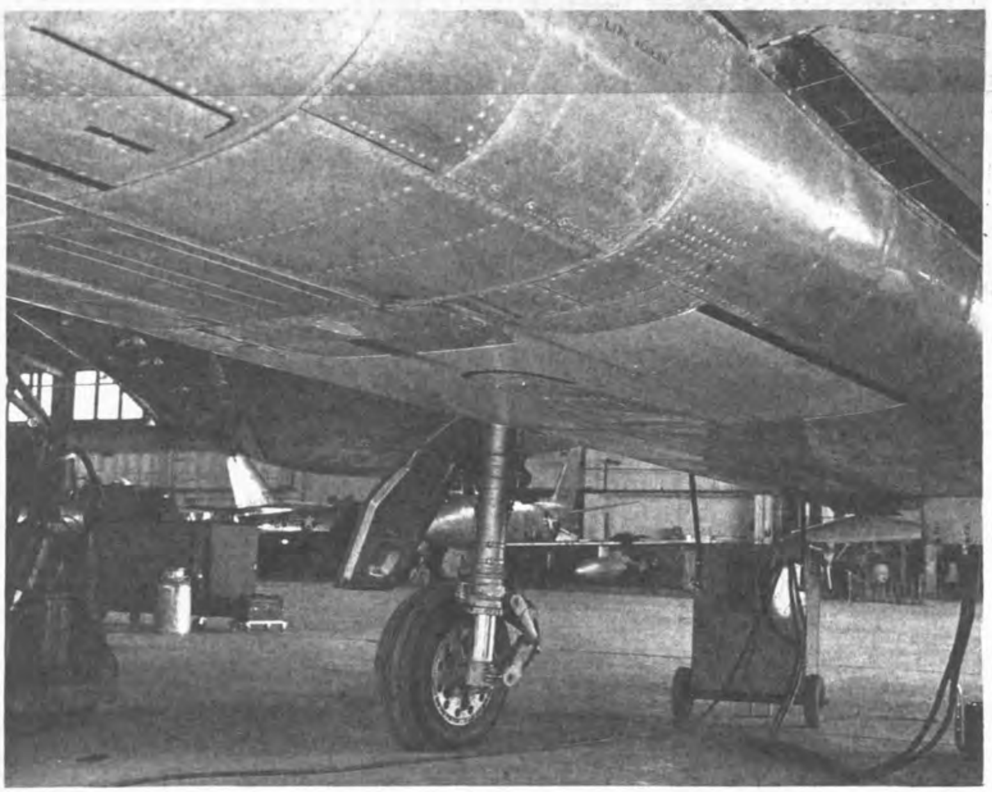


SLATS OPEN

SLATS CLOSED

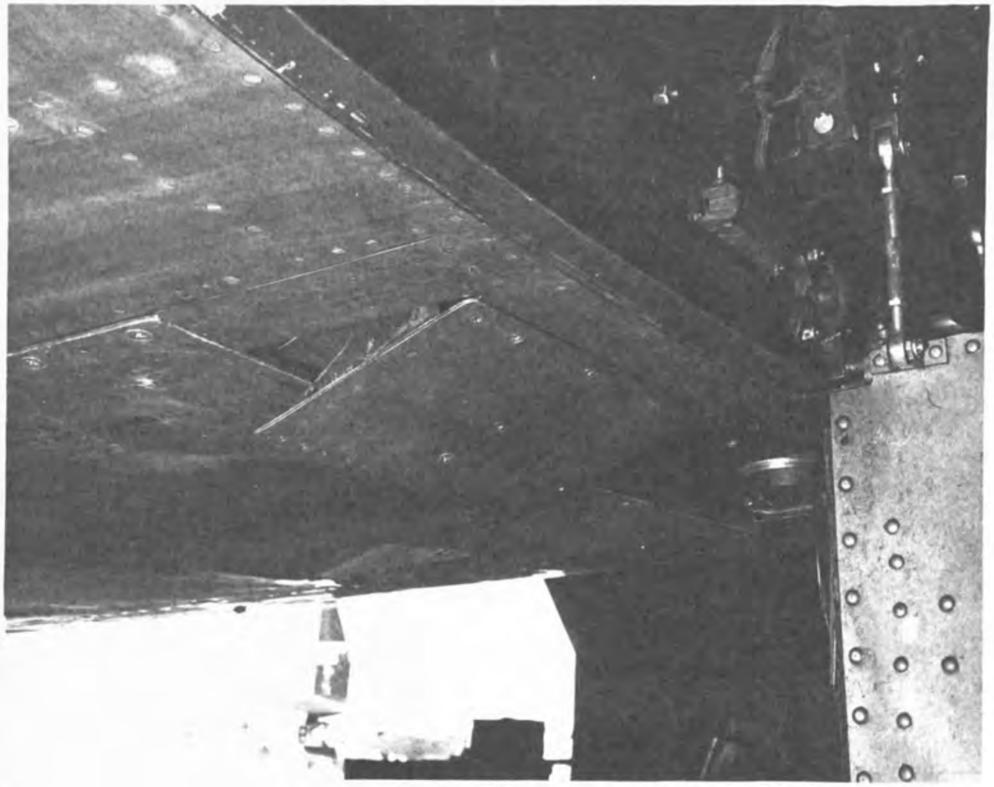


SPEED BRAKE OPEN

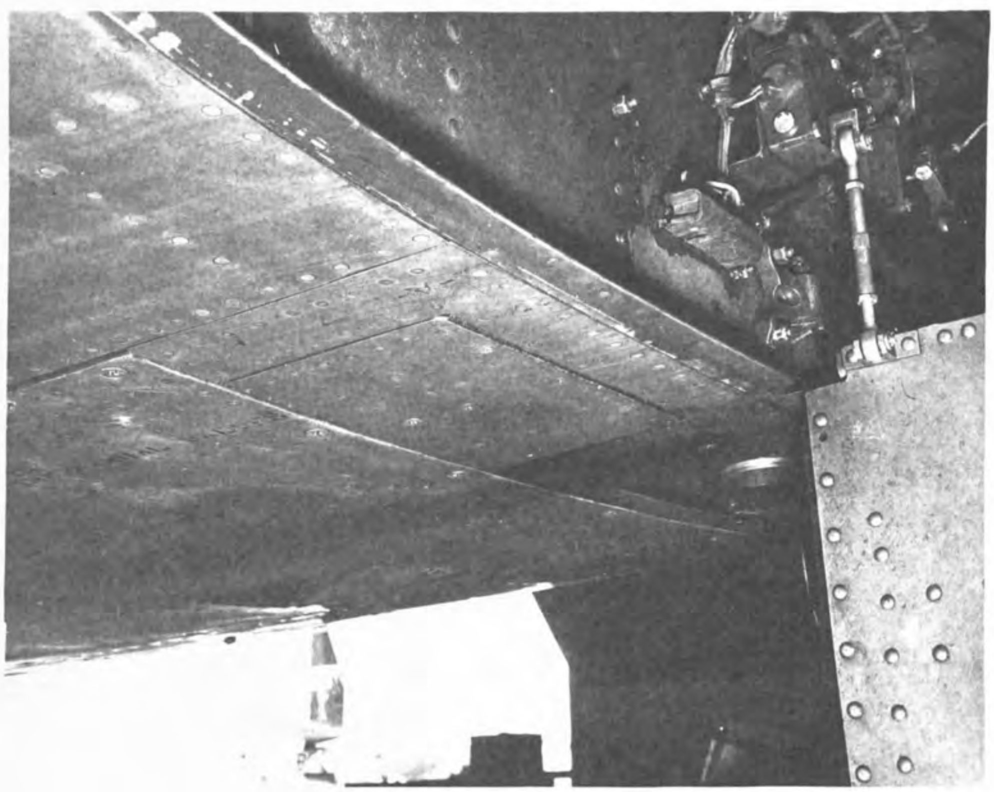


SPEED BRAKE CLOSED

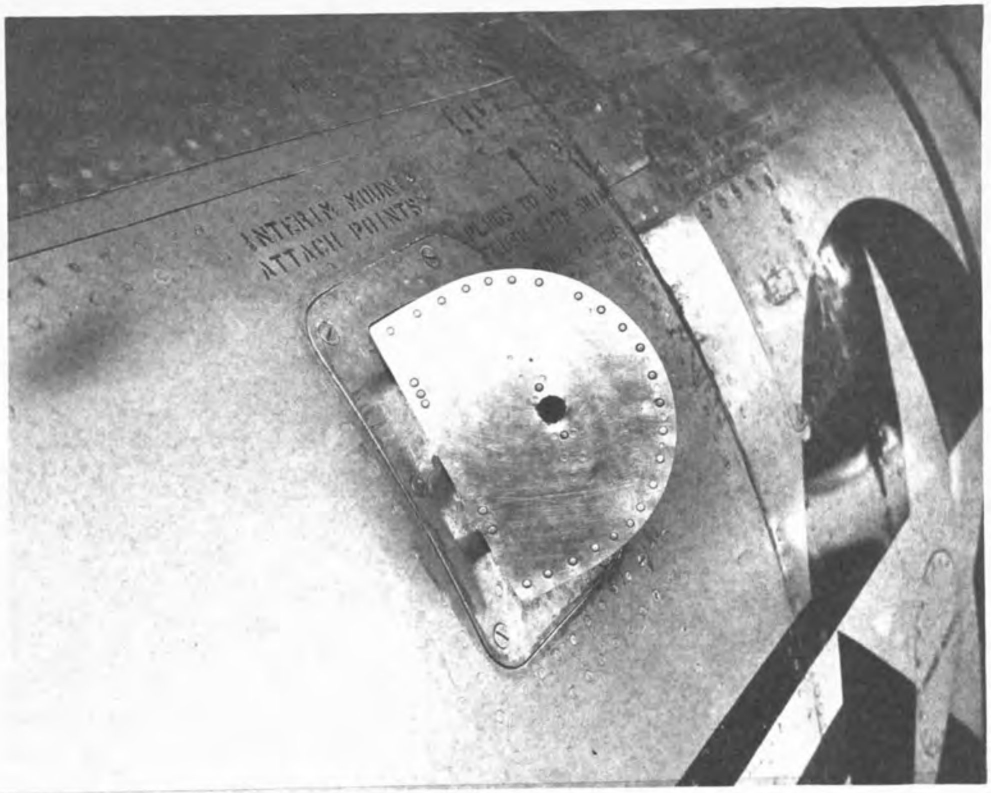
APPENDIX II



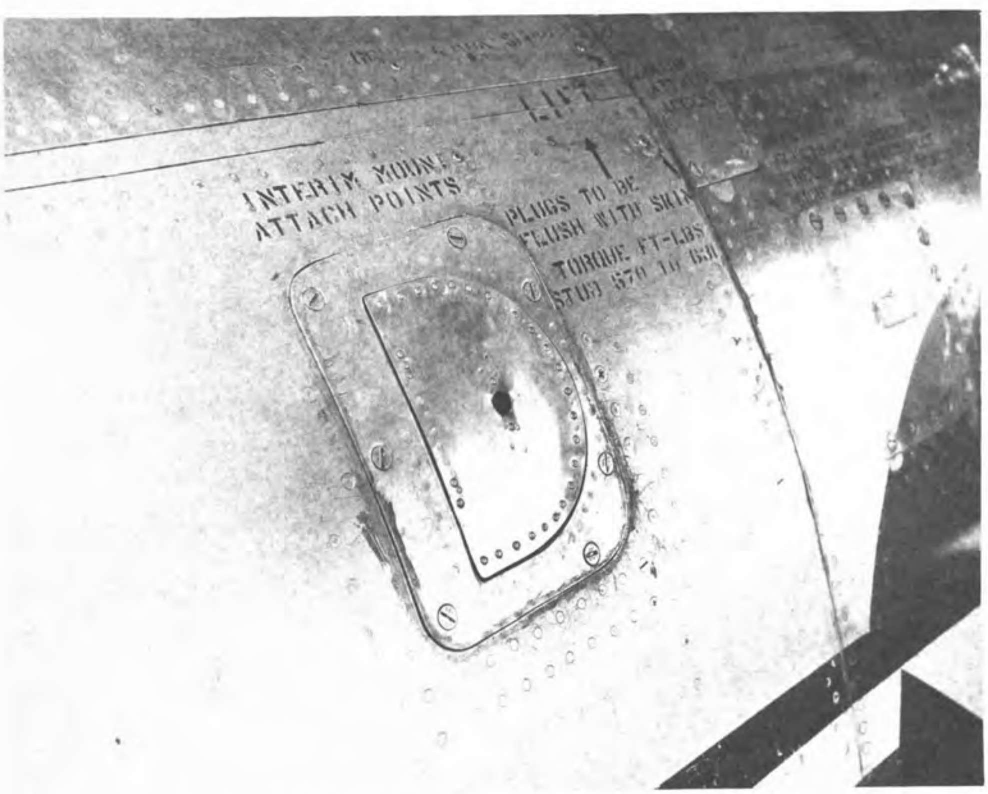
OIL COOLER DOOR OPEN



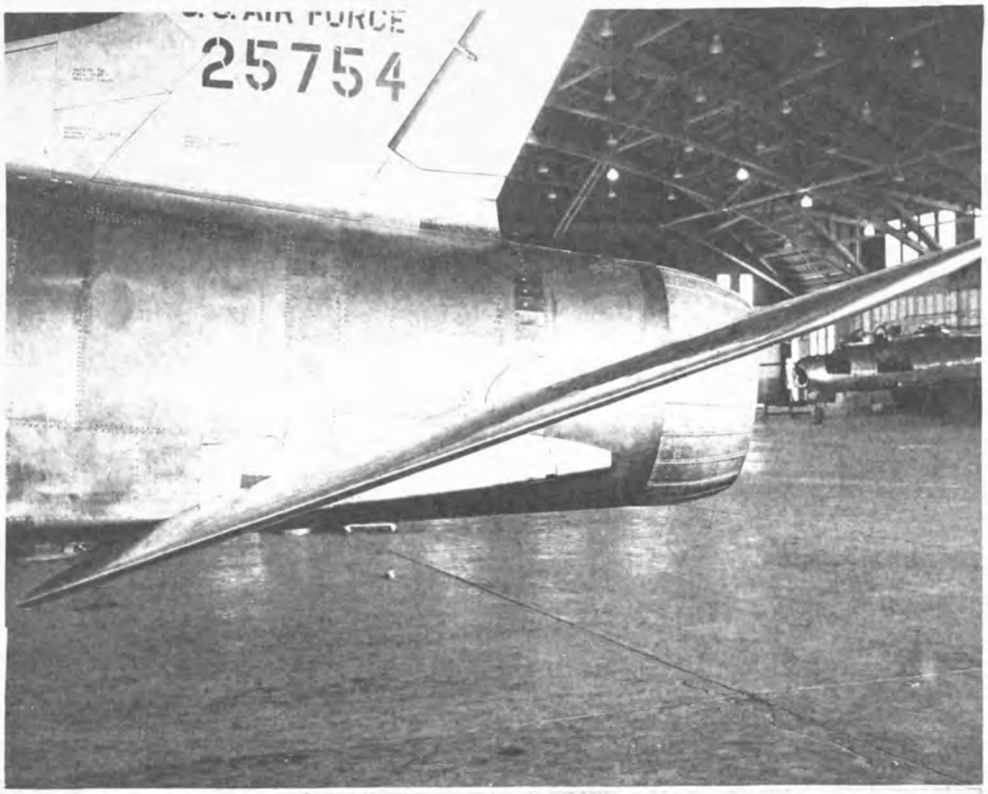
OIL COOLER DOOR CLOSED



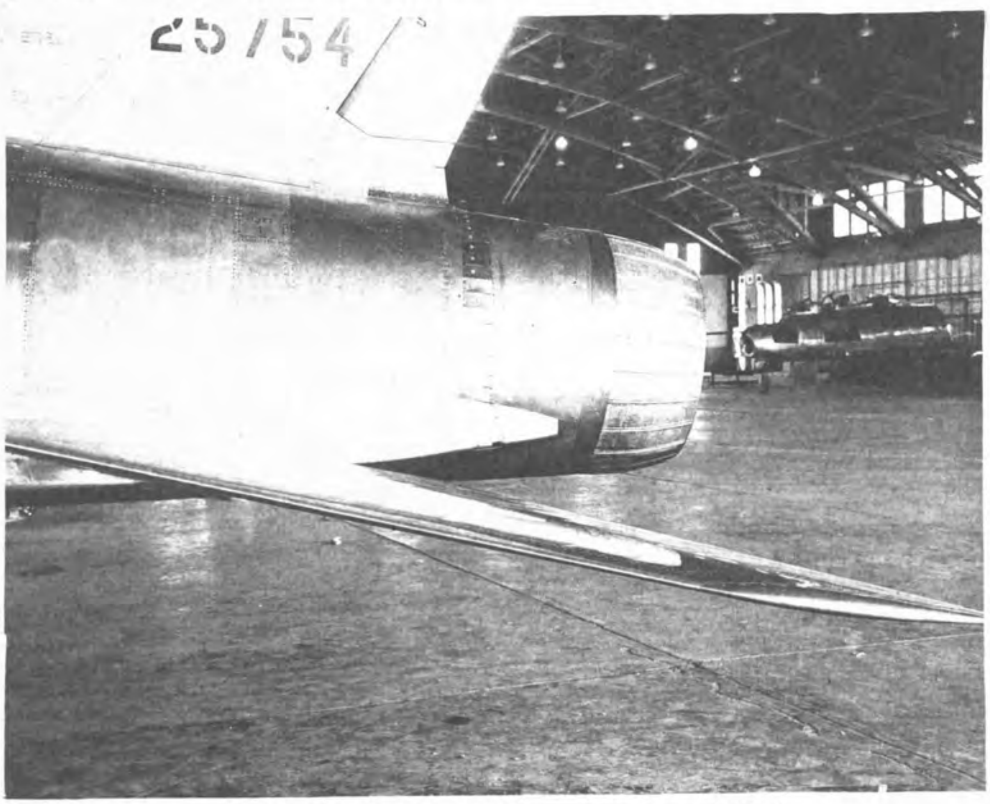
SURGE BLEED VALVE DOOR OPEN



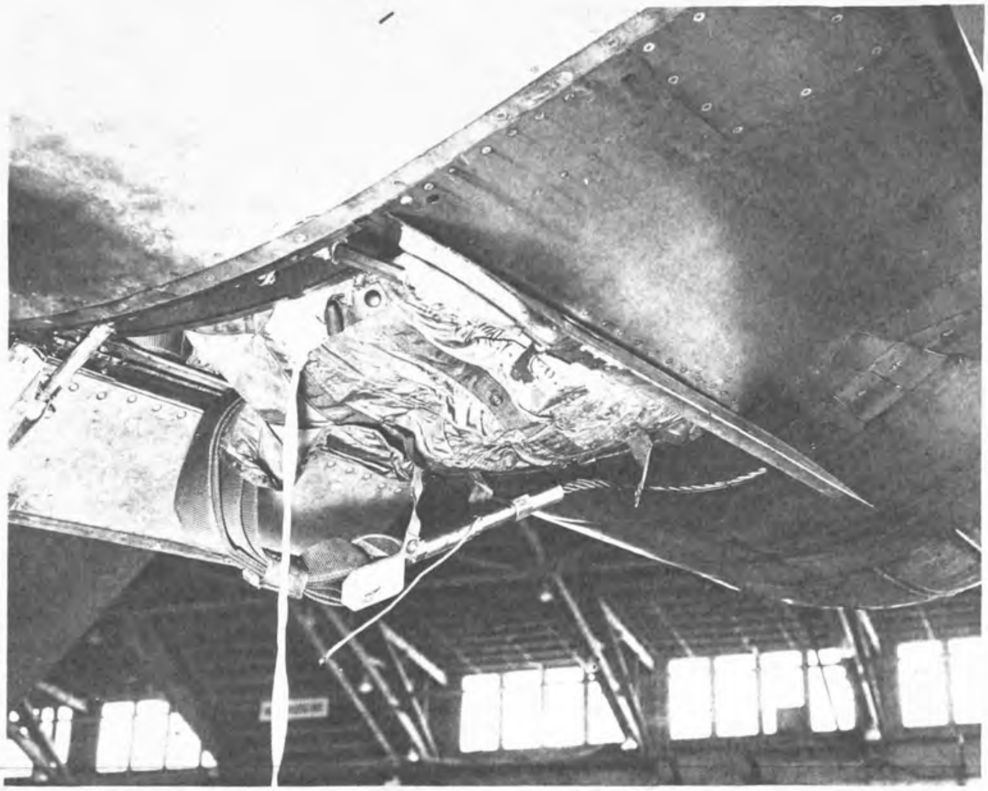
SURGE BLEED VALVE DOOR CLOSED



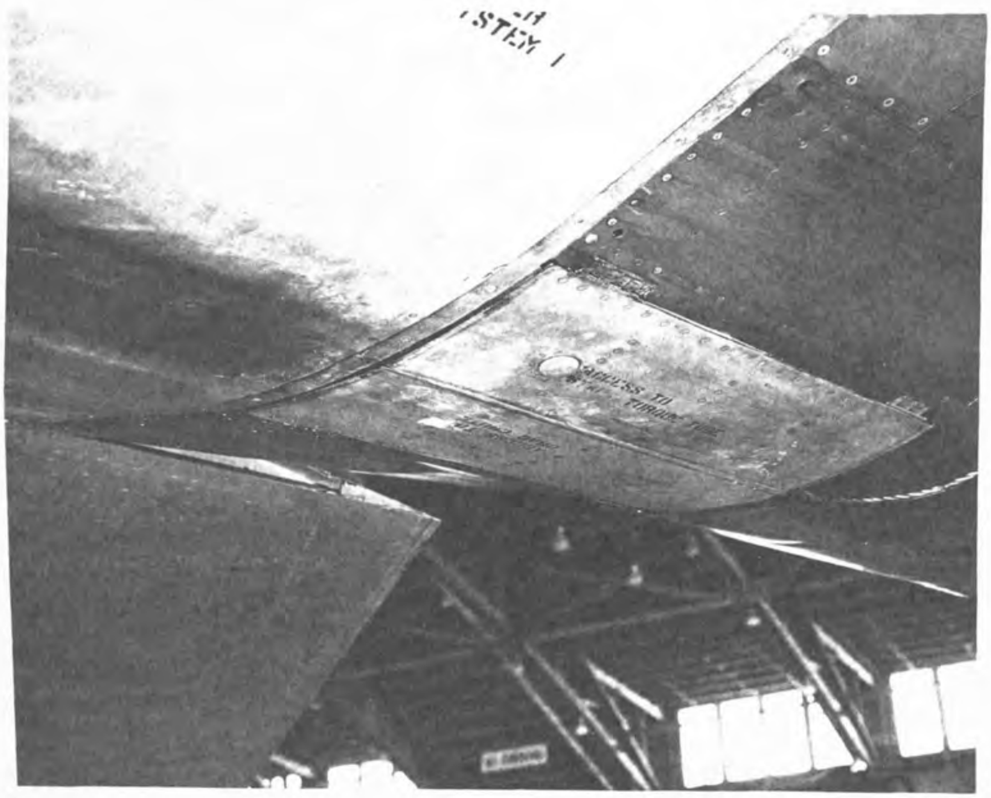
STABILIZER FULL NOSE UP



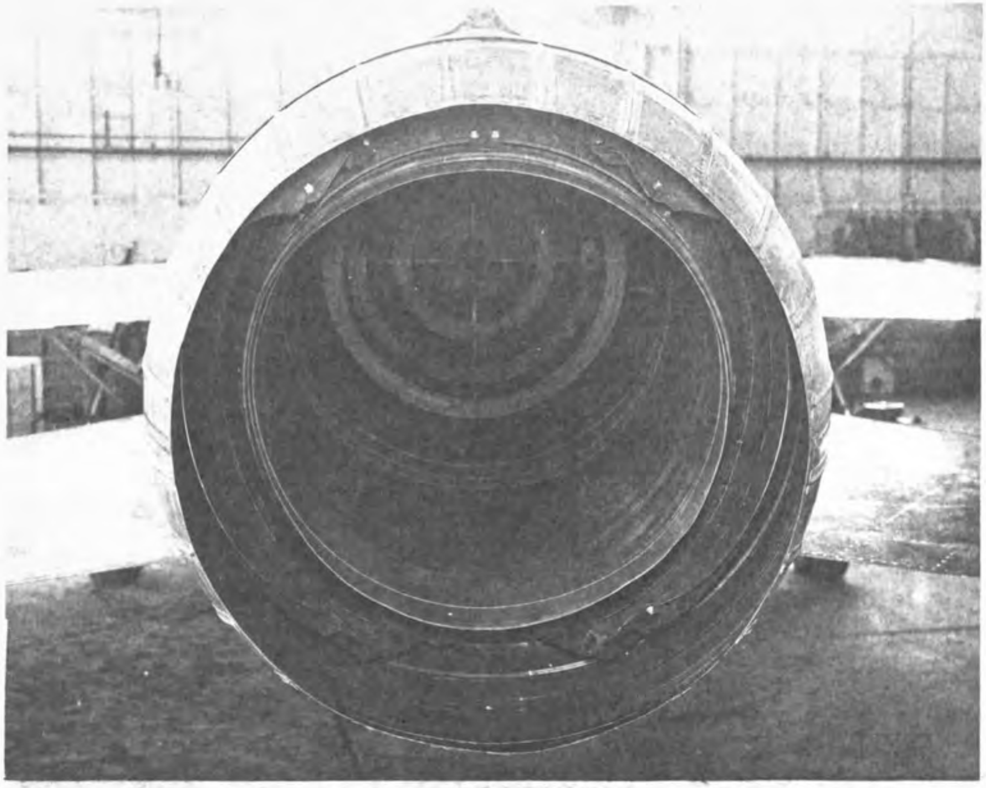
STABILIZER FULL NOSE DOWN



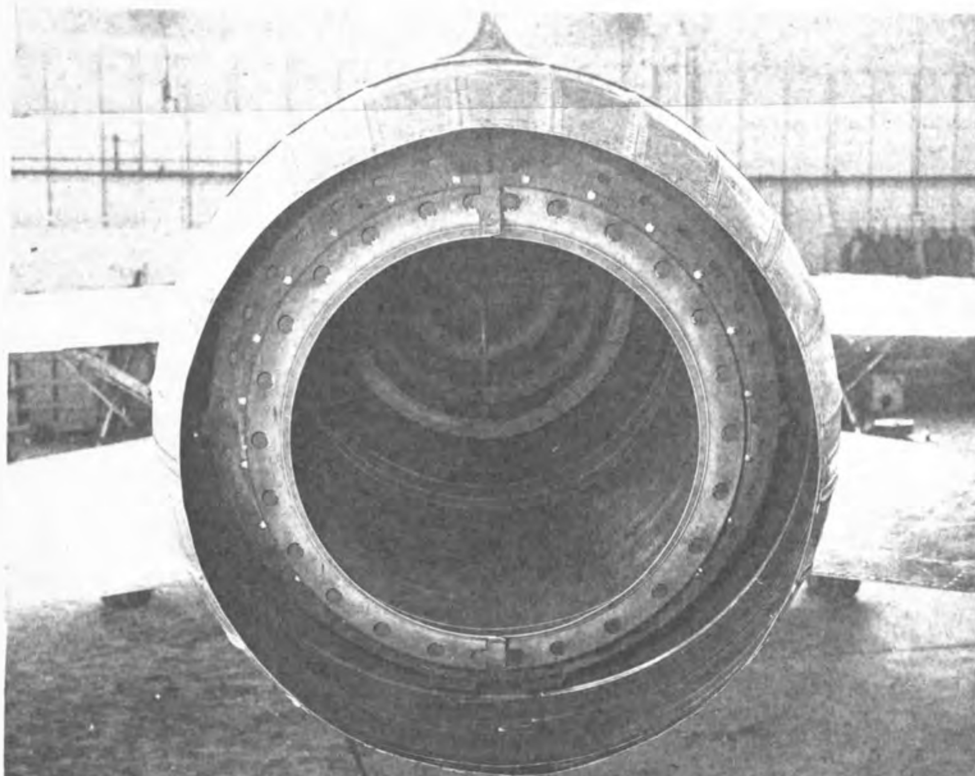
DRAG CHUTE DOOR OPEN



DRAG CHUTE DOOR CLOSED



EXHAUST NOZZLE OPEN



EXHAUST NOZZLE CLOSED

APPENDIX III

FLIGHT LOG AND ORIGINAL DATA
YF-100A, USAF No. 52-5754

FLIGHT LOG

Flight No.	Date	Flight Time	Pilot	Tests
1	9-3-53	:58	Captain Hopkins	Military power check climb, Max. level flight speed at 45,000 feet.
2	9-3-53	:38	Captain Hopkins	Max. power check climb, airspeed calibration at 40,000 feet.
3	9-4-53	:28	Captain Hopkins	Performance T.O., airspeed calibration by tower fly-bys.
4	9-4-53	:47	Captain Hopkins	Performance and stability T.O., Dynamic longitudinal, lateral and directional stability in the power and power approach configuration at 10,000 feet, max. C_L landing.
5	9-4-53	:40	Captain Hopkins	Performance and stability T.O., max. power climb, speed-power at 41,500 feet, airspeed calibration.
6	9-4-53	:17	Captain Hopkins	Flight aborted due to engine malfunction.
7	9-10-53	:54	Lt. Col. Everest	Military power check climb, speed-power at 41,500 feet, Max. speed points at 35,000, Stalls at 35,000 feet.
8	9-10-53	:30	Lt. Col. Everest	Stability T.O., Max. power climb, Maneuvering flight at 45,000 feet.
9	9-11-53	:30	Lt. Col. Everest	Performance and stability T.O., Maneuvering flight at 10,000 ft, stalls at 10,000 feet.
10	9-11-53	:22	Lt. Col. Everest	Performance and stability T.O., Speed-power at 11,500 feet.
11	9-11-53	:27	Lt. Col. Everest	Speed-power at 25,000 feet.
12	9-11-53	:23	Lt. Col. Everest	Stability T.O., max. speed at 35,000 feet, max. "g" supersonic diving turn at 35,000 feet, Static thrust run.

Flight No.	Date	Flight Time	Pilot	Tests
13	9-12-53	:20	Lt. Col. Everest	Maximum speed at 2500 feet, longitudinal trim change with dive brake opening, stability and performance landing.
14	9-12-53	:54	Lt. Col. Everest	Performance T.O., longitudinal trim change, performance landing, performance T.O. (Mil. power), performance landing.
15	9-12-53	:40	Lt. Col. Everest	Performance T.O., Comparative performance with F-86-E.
16	9-12-53	:42	Lt. Col. Everest	Speed-power at 25,000 ft, Stalls at 35,000 ft.
17	9-12-53	:31	Lt. Col. Everest	Max. power check climb, Dynamic longitudinal, Lateral and direction stability at 45,000 feet. Stalls at 45,000 feet.
18	9-13-53	:40	Captain Hopkins	Max. power climbs. Static longitudinal stability at 35,000 feet.
19	9-13-53	:34	Major Yeager	Familiarization flight.
20	9-14-53	:25	Lt. Col. Everest	Performance T.O., max. power climb. Max. speed at 50,000 feet. Dive.
21	9-14-53	:31	Lt. Col. Everest	Max. power climb. Dive.
22	9-14-53	:24	Major Murray	Familiarization flight.
23	9-14-53	:29	Lt. Col. Everest	Max. power climb. Aileron rolls at 45,000 ft. Dive.
24	9-14-53	:27	Lt. Col. Everest	Max. power climb. Max. speed at 51,000 ft. Dive. Performance landing.

Flight No.	Date	Flight Time	Pilot	Tests
25	9-14-53	:04	Lt. Col. Everest	Night flight--gear failed to retract--flight aborted.
26	9-15-53	:40	Major Stephens	Familiarization flight.
27	9-15-53	:30	Col. Hanes	Familiarization flight.
28	9-15-53	:19	Lt. Col. Everest	Performance T.O. Side-slips at 10,000 ft. Performance landing.
29	9-15-53	:13	Lt. Col. Everest	Performance T.O. Aileron rolls at 10,000 ft. Performance landing.
30	9-15-53	:03	Lt. Col. Everest	Performance T.O. Performance landing.
31	9-15-53	:28	Lt. Col. Everest	Night flight evaluation. Level flight acceleration at 20,000 and 35,000 ft.
32	9-16-53	:29	Lt. Col. Everest	Performance T.O. Aileron rolls at 10,000 ft.
33	9-16-53	:27	Lt. Col. Everest	Static longitudinal stability at 10,000 feet.
34	9-16-53	:23	Lt. Col. Everest	Max. speed at 50,000 ft. Dive.
35	9-17-53	:45	Maj. Gen. Boyd	Evaluation flight.
36	9-17-53	:28	Lt. Col. Everest	Level flight acceleration at 20,000 and 35,000 feet. Static longitudinal stability at 10,000 feet.
37	9-17-53	:20	Lt. Col. Everest	Evaluation of external tank installation.
38	9-17-53	:25	Captain Hopkins	Speed-power at 25,000 ft. with external tanks.
39	9-17-53	:57	Captain Hopkins	Speed-power at 25,000 ft. with external tanks.
	TOTAL	19:42		

DATA CORRECTED FOR INSTRUMENT ERROR
YF-100 USAF No. 52-5754

Test	STATIC THRUST RUN								
Flight No.	—	—	—	—	—	—	—	—	—
Run No.	1 *	2	3	4	5	6	7	8	9
Altitude - Ft	—	—	—	—	—	—	—	—	—
IAS - Knots	—	—	—	—	—	—	—	—	—
OAT - °C	21	21	20.6	20.6	20.6	20	20	20	20
RPM (high press)	9480	9480	9405	9215	9025	8600	8280	7630	6825
RPM (low press)	5770	5690	5360	5320	5045	4365	3900	3120	2530
Exhaust Gas Temp - °C	562	552	542	462	430	405	367	275	265
Fuel Used (Eng) - Gal	—	—	—	—	—	—	—	—	—
Fuel Used (A/B) - Gal	—	—	—	—	—	—	—	—	—
Fuel Flow (total) - GPH	3158	818	776	670	552	377	280	187	143.4
Oil Cooler GAP - DEG	9.4	9.4	9.4	9.4	9.4	9.4	9.4	9.4	9.4
Bleed Valve Position	C	C	C	C	C	O	O	C	C
Fuel Weight - lbs/gal	6.48	6.48	6.48	6.48	6.48	6.48	6.48	6.48	6.48
Nozzle Position	O	C	C	C	C	C	C	C	C
Time - Sec	—	—	—	—	—	—	—	—	—
GROSS THRUST - LBS	9045	5788	5408	4685	3937	2507	1917	1250	758
BARO. PRESS - IN Hg	27.597	27.597	27.597	27.597	27.597	27.597	27.597	27.597	27.597

* A/B ON

Test	STATIC THRUST RUN								
Flight No.	—	—	—	—	—	—	—	—	—
Run No.	10	11	12	13	14	15	16	17	18
Altitude - Ft	—	—	—	—	—	—	—	—	—
IAS - Knots	—	—	—	—	—	—	—	—	—
OAT - °C	20	20	20	20	20	20	20	20	20
RPM (high press)	6050	5680	5675	6020	6845	7610	8230	8620	9000
RPM (low press)	2175	1980	1915	2170	2695	3280	3870	4410	5020
Exhaust Gas Temp - °C	265	265	250	265	265	278	355	425	430
Fuel Used (Eng) - Gal	—	—	—	—	—	—	—	—	—
Fuel Used (A/B) - Gal	—	—	—	—	—	—	—	—	—
Fuel Flow (total) - GPH	124.9	109.5	110.1	150.5	153.8	202	299.5	395	558
Oil Cooler GAP - DEG	9.4	9.4	9.4	9.4	9.4	9.4	9.4	9.4	9.4
Bleed Valve Position	C	C	C	C	C	C	O	O	C
Fuel Weight - lbs/gal	6.48	6.48	6.48	6.48	6.48	6.48	6.48	6.48	6.48
Nozzle Position	C	C	O	C	C	C	C	C	C
Time - Sec	—	—	—	—	—	—	—	—	—
GROSS THRUST - LBS	465	382	220	475	783	1240	1848	2540	3857
BARO. PRESS - IN Hg	27.597	27.597	27.597	27.597	27.597	27.597	27.597	27.597	27.597

Test	STATIC THRUST RUN								
Flight No.	—	—	—	—	—	—	—	—	—
Run No.	19	20	21	22*	23 *	24	25	26	27
Altitude - Ft	—	—	—	—	—	—	—	—	—
IAS - Knots	—	—	—	—	—	—	—	—	—
OAT - °C	20	20	203	20.3	20.3				
RPM (high press)	9240	9420	9510	9540	9020				
RPM (low press)	5290	5490	5760	5740	4960				
Exhaust Gas Temp - °C	500	540	558	558	448				
Fuel Used (Eng) - Gal	—	—	—	—	—	—	—	—	—
Fuel Used (A/B) - Gal	—	—	—	—	—	—	—	—	—
Fuel Flow (total) - GPH	663	740	809.5	3045	2350				
Oil Cooler GAP - DEG	9.4	9.4	9.4	9.4	9.4				
Bleed Valve Position	C	C	C	C	C				
Fuel Weight - lbs/gal	6.48	6.48	6.48	6.48	6.48				
Nozzle Position	C	C	C	O	O				
Time - Sec	—	—	—	—	—	—	—	—	—
GROSS THRUST - LBS	4630	5265	5700	8745	6035				
BARO. PRESS - IN Hg	27.597	27.597	27.597	27.597	27.597				

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Test	AIRSPEED CALIBRATION (TOWER FLY-BY)								
Flight No.	3	3	3	3	3	3	3	3	
Run No.	1	2	3	4	5	6	7	8	
Altitude - Ft	1950	2105	2200	2260	2300	1830	2140	2150	
IAS - Knots	600	495	412.5	301.5	250	600	565	452	
OAT - °C									
RPM (high press)									
RPM (low press)									
Exhaust Gas Temp - °C									
Fuel Used (Eng) - Gal									
Fuel Used (A/B) - Gal									
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.									
Bleed Valve Position									
Fuel Weight - lbs/gal									
Nozzle Position									
Time - Sec									
PRESSURE ALT-TRUE - FT	2507	2409	2527	2461	2455	2448	2540	2510	
AVG GROSS WT LBS	← 23000 LBS →								
GEAR POS	UP	UP	UP	UP	UP	UP	UP	UP	

Test	AIRSPEED CALIBRATION (AIRPLANE FLY-BY)								
Flight No.	2	2	2	2	2				
Run No.	1	2	3	4	5				
Altitude - Ft	38785	39160	39355	39425	39415				
IAS - Knots	304.5	296	274.5	262.5	192.5				
OAT - °C	—	-15°	-19°	-22°	-35.5°				
RPM (high press)									
RPM (low press)									
Exhaust Gas Temp - °C									
Fuel Used (Eng) - Gal									
Fuel Used (A/B) - Gal									
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.									
Bleed Valve Position									
Fuel Weight - lbs/gal									
Nozzle Position									
~~Time - Sec~~ AVG GR WT	← 23000 LBS →								
PACER CAS MPH	261.5	249	261.5	264.5	226.5				
PACER ALT FT	39865	39865	39870	39860	39705				

Test	AIRSPEED CALIBRATION (PACER)								
Flight No.	2	5							
Run No.	5	1							
Altitude - Ft	39415	227							
IAS - Knots	192.5	42360							
OAT - °C	-35.5	-37							
RPM (high press)									
RPM (low press)									
Exhaust Gas Temp - °C									
Fuel Used (Eng) - Gal									
Fuel Used (A/B) - Gal									
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.									
Bleed Valve Position									
Fuel Weight - lbs/gal									
Nozzle Position									
~~Time - Sec~~ AVG GR.WT.	23000 LBS								
PACER CAS MPH	226.5	265.5							
PACER ALT FT	39205	42555							

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Test	LEVEL FLIGHT SPEED POWER (CLEAN)								
Flight No.	13	13		10	10	10	10	14	14
Run No.	2	3		1	2	3	4	1	2
Altitude - Ft	2545	2545		9975	12265	13460	14190	10200	11000
IAS - Knots	591.5	470		533	492	463.5	418	444.5	392.5
OAT - °C	83	64		66	54.5	47	38	52	41.5
RPM (high press)	9545	9550		9530	8940	9510	9255	9440	9190
RPM (low press)	5350	5500		5440	4815	5615	5365	5485	5270
Exhaust Gas Temp - °C	455	475		500	410	510	450	475	425
Fuel Used (Eng) - Gal									
Fuel Used (A/B) - Gal									
Fuel Flow (total) - GPH	3560	829		3273	2170	764	578	696	561
Oil Cooler GAP - in.	9.6	9.6		9.5	0.25	9.45	0.2	9.4	9.5
Bleed Valve Position	C	C		C	C	C	C	C	C
Fuel Weight - lbs/gal	6.47	6.47		6.55	6.55	6.55	6.55	6.48	6.48
Nozzle Position	O	C		O	O	C	C	C	C
Time - Sec									
Gross Weight ~ lbs.	23050	22380		22810	21630	21180	20940	23840	23660

Test	LEVEL FLIGHT SPEED POWER (CLEAN)								
Flight No.	14	14	14	14	14	14	14	14	
Run No.	3	4	5	6	7	8	9	10	
Altitude - Ft	11600	12190	12490	12220	12060	12000	12000	12200	
IAS - Knots	359	309	272	237	192.5	160	141.5	956	
OAT - °C	92.5	26.5	22.0	18.5	15	—	13	—	
RPM (high press)	9100	8900	8820	8815	8760	9020	9060	9010	
RPM (low press)	5235	5060	5005	5005	4950	—	5285	—	
Exhaust Gas Temp - °C	410	375	375	375	375	—	450	—	
Fuel Used (Eng) - Gal									
Fuel Used (A/B) - Gal									
Fuel Flow (total) - GPH	514	423	397	370.5	368	—	493.5	—	
Oil Cooler GAP - in.	0.7	1.3	1.8	3.15	5.2	—	5.95	—	
Bleed Valve Position	C	O	O	O	O	C	C	C	
Fuel Weight - lbs/gal	6.48	6.48	6.48	6.48	6.48	6.48	6.48	6.48	
Nozzle Position	C	C	C	C	C	C	C	C	
Time - Sec									
Gross Weight ~ lbs.	23560	23430	23300	23200	23040	22900	22780	22600	

Test	LEVEL FLIGHT SPEED POWER (CLEAN)								
Flight No.	11	11	11	11	11	11	16	16	16
Run No.	1	2	3	4	5	6	1	3	4
Altitude - Ft	26340	27120	27700	28280	28680	28780	27150	27890	27910
IAS - Knots	403.5	390.5	381	370	347	318	289	228	185.5
OAT - °C	22.5	20	—	15.5	10.5	4.1	-0.5	—	-18.5
RPM (high press)	9510	9140	8700	9475	9040	9025	8750	8750	8980
RPM (low press)	5770	5335	—	5840	5375	5365	—	—	5540
Exhaust Gas Temp - °C	545	475	—	535	460	450	425	—	495
Fuel Used (Eng) - Gal									
Fuel Used (A/B) - Gal									
Fuel Flow (total) - GPH	2438	1946	—	584	393.3	370.5	320.8	—	387
Oil Cooler GAP - in.	9.7	0.6	—	1.65	1.6	3.9	2.39	—	5.3
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal	6.47	6.47	6.47	6.47	6.47	6.47	6.43	6.43	6.43
Nozzle Position	O	O	—	C	C	C	C	C	C
Time - Sec									
Gross Weight ~ lbs.	23380	21930	21670	21420	21270	21140	23390	23100	23010

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Test	LEVEL FLIGHT SPEED POWER (CLEAN)								
Flight No.	16	16		7	7	12	12	22	
Run No.	5	6		2	3	1	2	1	
Altitude - Ft	27740	27940		34700	34900	34130	34850	33710	
IAS - Knots	155	292.5		335	314	341.5	317.5	344.5	
OAT - °C	-21.5	-3.0		-2.0	-7.5	0.5	-6.0	0	
RPM (high press)	8990	8730		9215	9120	9330	9200	9260	
RPM (low press)	5565	5165		5655	5710	5735	5735	5670	
Exhaust Gas Temp - °C	495	415		525	500	530	510	525	
Fuel Used (Eng) - Gal									
Fuel Used (A/B) - Gal									
Fuel Flow (total) - GPH	364.0	319		1805	424.8	1990	398	1856	
Oil Cooler GAP - in.	5.3	3.4		0.2	0.6	1.4	2.2	0	
Bleed Valve Position	C	C		C	C	C	C	C	
Fuel Weight - lbs/gal	6.43	6.43		6.425	6.425	6.48	6.48	6.42	
Nozzle Position	C	C		O	C	O	C	O	
Time - Sec									
Gross Weight ~ lbs.	22860	22450		22420	22200	22180	21900	22050	

Test	LEVEL FLIGHT SPEED POWER (CLEAN)								
Flight No.	5	5	5	5	7		1	1	
Run No.	1	2	3	4	1		1	2	
Altitude - Ft	40920	42330	42770	43160	40870		44530	44630	
IAS - Knots	290.5	273.5	245.5	226.5	203.5		265.5	244	
OAT - °C	-15.5	-21	-28.5	-35	-37		-22	—	
RPM (high press)	9135	8590	8970	8850	8890		9030	8955	
RPM (low press)	5090	5065	5730	5655	5700		5660	—	
Exhaust Gas Temp - °C	545	450	520	500	510		530	—	
Fuel Used (Eng) - Gal									
Fuel Used (A/B) - Gal									
Fuel Flow (total) - GPH	1380	1075	302	278.6	290		1320	—	
Oil Cooler GAP - in.	9.7	1.65	2.6	5.3	5.3		5.3	—	
Bleed Valve Position	C	C	C	C	C		C	C	
Fuel Weight - lbs/gal	6.38	6.38	6.38	6.38	6.425		6.38	6.38	
Nozzle Position	O	O	C	C	C		O	C	
Time - Sec									
Gross Weight ~ lbs.	22270	21880	21630	21340	22920		21950	21600	

Test	LEVEL FLIGHT SPEED POWER (CLEAN)								
Flight No.	20								
Run No.	1								
Altitude - Ft	49550								
IAS - Knots	231.5								
OAT - °C	-28								
RPM (high press)	8970								
RPM (low press)	5695								
Exhaust Gas Temp - °C	570								
Fuel Used (Eng) - Gal									
Fuel Used (A/B) - Gal									
Fuel Flow (total) - GPH	1032								
Oil Cooler GAP - in.	5.3								
Bleed Valve Position	C								
Fuel Weight - lbs/gal	6.465								
Nozzle Position	O								
Time - Sec									
Gross Weight ~ lbs.	22060								

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Test	LEVEL FLIGHT SPEED POWER (TANKS)								
Flight No.	38	38	38	38	39	39	39	39	39
Run No.	1	2	3	4	1	2	3	4	5
Altitude - Ft	26180	27050	27700	27840	25900	26510	26790	27290	27645
IAS - Knots	394	385	368.5	356	329	296	250	205	179
OAT - °C	16	14	10	6.5	2	-3	-10	-18	-22
RPM (high press)	9465	9260	8750	9300	9040	8885	8785	8745	8845
RPM (low press)	5725	5530	5130	5725	5310	5275	5280	5200	5425
Exhaust Gas Temp - °C	530	500	400	500	455	425	425	430	450
Fuel Used (Eng) - Gal									
Fuel Used (A/B) - Gal									
Fuel Flow (total) - GPH	2390	2078	1471	5645	454	382.9	350.5	328.9	361
Oil Cooler GAP - in.	9.4	9.4	0.9	0.25	0.1	0.1	0.6	1.22	2.4
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal	6.47	6.47	6.47	6.47	6.38	6.38	6.38	6.38	6.38
Nozzle Position	O	O	O	C	C	C	C	C	C
Time - Sec									
Gross Weight ~ lbs	22900	22200	21730	21480	23460	22960	22580	22170	28050

Test									
Flight No.									
Run No.									
Altitude - Ft									
IAS - Knots									
OAT - °C									
RPM (high press)									
RPM (low press)									
Exhaust Gas Temp - °C									
Fuel Used (Eng) - Gal									
Fuel Used (A/B) - Gal									
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.									
Bleed Valve Position									
Fuel Weight - lbs/gal									
Nozzle Position									
Time - Sec									

Test									
Flight No.									
Run No.									
Altitude - Ft									
IAS - Knots									
OAT - °C									
RPM (high press)									
RPM (low press)									
Exhaust Gas Temp - °C									
Fuel Used (Eng) - Gal									
Fuel Used (A/B) - Gal									
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.									
Bleed Valve Position									
Fuel Weight - lbs/gal									
Nozzle Position									
Time - Sec									

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Test	CHECK CLIMB ~ MAX. POWER								
Flight No.	2	2	2	2	2	2	2	2	2
Run No.									
Altitude - Ft	3960	4935	5990	6980	8030	8965	9970	10960	11960
IAS - Knots	469.0	481.0	479.0	469.0	460.5	454.0	454.5	453.5	446.5
OAT - °C	65.5	65.5	63.9	60.0	56.0	53.5	51.5	50.0	49.5
RPM (high press)	9530	9530	9540	9530	9530	9530	9530	9525	9530
RPM (low press)	5965	5460	5480	5485	5505	5530	5540	5550	5560
Exhaust Gas Temp - °C	490	490	495	515	520	520	525	525	530
Fuel Used (Eng) - Gal	37.6	41.6	43.5	45.2	46.6	48.0	49.7	51.2	52.6
Fuel Used (A/B) - Gal	99.4	112.2	118.4	124.0	128.4	132.4	138.0	143.0	147.0
Fuel Flow (total) - GPH									
Oil Cooler GAP - in °C	0.2	0.2	0.2	0.2	0.2	0.2	0.2	0.2	0.2
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	O	O	O	O	O	O	O	O	O
Time - Sec	0	17.9	26.7	34.2	41.0	47.1	55.0	62.7	69.5

Test	CHECK CLIMB ~ MAX. POWER								
Flight No.	2	2	2	2	2	2	2	2	2
Run No.									
Altitude - Ft	12940	13995	15020	15990	16960	18010	18980	19945	20945
IAS - Knots	444.5	435.5	427.0	419.0	416.0	413.0	407.0	416.0	399.5
OAT - °C	47.0	44.0	44.0	41.5	38.5	36.0	34.0	32.5	30.0
RPM (high press)	9530	9540	9545	9540	9540	9540	9535	9530	9530
RPM (low press)	5570	5600	5615	5630	5645	5660	5670	5680	5740
Exhaust Gas Temp - °C	535	540	545	545	545	555	560	560	555
Fuel Used (Eng) - Gal	53.8	54.9	56.1	57.1	58.3	59.6	60.6	61.9	62.8
Fuel Used (A/B) - Gal	151.0	154.0	157.4	160.4	164.0	168.0	171.0	174.4	177.4
Fuel Flow (total) - GPH									
Oil Cooler GAP - in °C	0.2	0.2	0.2	0.2	0.1	0.1	0.1	0.1	0.1
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	O	O	O	O	O	O	O	O	O
Time - Sec	74.7	80.2	85.5	90.6	96.8	103.2	109.0	115.0	120.5

Test	CHECK CLIMB ~ MAX POWER								
Flight No.	2	2	2	2	2	2	2	2	2
Run No.									
Altitude - Ft	21970	22950	24015	24990	25985	26930	27880	28940	29975
IAS - Knots	394.5	391.0	383.0	375.5	370.5	368.0	362.5	356.5	351.0
OAT - °C	26.5	24.5	21.5	18.0	16.0	13.0	10.5	7.0	4.0
RPM (high press)	9520	9520	9520	9520	9520	9520	9520	9525	9520
RPM (low press)	5720	5750	5780	5795	5810	5840	5860	5890	5910
Exhaust Gas Temp - °C	550	545	545	545	560	560	560	565	565
Fuel Used (Eng) - Gal	64.0	65.1	66.1	67.1	68.1	69.2	70.0	71.0	72.0
Fuel Used (A/B) - Gal	181.0	184.0	187.0	189.0	192.4	195.4	198.0	200.4	203
Fuel Flow (total) - GPH									
Oil Cooler GAP - in °C	0.1	0.1	0.1	0.1	0.1	0.1	0.1	0.1	0.1
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	O	O	O	O	O	O	O	O	O
Time - Sec	126.7	132.7	138.0	143.3	149.4	155.2	160.2	165.5	171.2

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Test	CHECK CLIMB ~ MAX. POWER								
Flight No.	2	2	2	2	2	2	2	2	2
Run No.									
Altitude - Ft	30825	31800	32750	33695	34710	35675	36635	37560	38480
IAS - Knots	343.0	325.5	323.0	324.0	316.5	308.0	303.5	299.5	293.0
OAT - °C	1.0	-2.0	-4.0	-6.0	-9.0	-10.5	-13.5	-14.0	-14.0
RPM (high press)	9510	9510	9445	9420	9405	9370	9330	9320	9295
RPM (low press)	5920	5910	5890	5900	5900	5875	5865	5860	5840
Exhaust Gas Temp - °C	565	565	565	565	565	565	565	560	560
Fuel Used (Eng) - Gal	73.0	73.6	74.6	75.5	76.3	77.1	78.0	79.2	80.0
Fuel Used (A/B) - Gal	205.0	208.0	210.4	212.8	215.2	217.4	220.0	224.0	226.4
Fuel Flow (total) - GPH									
Oil Cooler GAP - °C	0.1	0.1	0.1	0.1	0.35	0.35	0.45	1.4	1.5
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	O	O	O	O	O	O	O	O	O
Time - Sec	176.0	181.6	187.7	193.9	199.8	205.2	212.2	222.3	230.0

Test	CHECK CLIMB ~ MAX POWER								
Flight No.	2	2	2			20	20	20	20
Run No.									
Altitude - Ft	39470	40435	41405			3090	3925	4965	5895
IAS - Knots	279.5	272.5	262.5			484.5	489.0	483.0	475.5
OAT - °C	-17.0	-20.0	-22.5			66	68	64.5	62
RPM (high press)	9295	9270	9265			9510	9510	9510	9505
RPM (low press)	5860	5870	5930			5490	5445	5465	5485
Exhaust Gas Temp - °C	560	565	565			490	495	495	510
Fuel Used (Eng) - Gal	80.8	81.4	82.4			74.3	76.3	77.8	79.0
Fuel Used (A/B) - Gal	229.0	230.4	231.0			93.0	99.2	104.2	108.0
Fuel Flow (total) - GPH									
Oil Cooler GAP - °C	2.3	2.9	3.5			0.1	0.2	0.2	0.2
Bleed Valve Position	C	C	C			C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	O	O	O			O	O	O	O
Time - Sec	236.5	242.5	252.0			0	13.5	20.8	26

Test	CHECK CLIMB ~ MAX POWER								
Flight No.	20	20	20	20	20	20	20	20	20
Run No.									
Altitude - Ft	6950	7925	8930	9930	10915	11935	12965	13950	14915
IAS - Knots	461.0	453.0	452.0	451.5	447.5	443.0	439.5	434.0	430.0
OAT - °C	58	56	53	51.5	49	46	43	40	37.5
RPM (high press)	9500	9500	9495	9490	9485	9480	9470	9480	9480
RPM (low press)	5455	5485	5485	5515	5520	5530	5540	5565	5580
Exhaust Gas Temp - °C	515	515	520	520	520	520	525	527	535
Fuel Used (Eng) - Gal	80.3	81.6	83.4	85.0	86.6	88.1	89.7	91.0	92.5
Fuel Used (A/B) - Gal	112.0	116.4	121.6	127.0	131.8	136.6	141.2	145.2	149.2
Fuel Flow (total) - GPH									
Oil Cooler GAP - in	0.1	0.1	0.1	0.1	0.1	0.1	0.1	0.1	0.1
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	O	O	O	O	O	O	O	O	O
Time - Sec	32	38.5	47	55.1	62.9	70.5	78.5	85	71

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Test	CHECK CLIMB ~ MAX POWER								
Flight No.	20	20	20	20	20	20	20	20	20
Run No.									
Altitude - Ft	15920	16980	17970	19015	20055	20990	22010	23015	24030
IAS - Knots	426.0	421.0	417.0	413.0	403.0	401.0	394.0	385.5	378.0
OAT - °C	34	31	29	28.5	26	25	23	20	17
RPM (high press)	9500	9480	9480	9480	9480	9470	9480	9470	9470
RPM (low press)	5605	5635	5650	5665	5685	5690	5710	5726	5755
Exhaust Gas Temp - °C	540	540	540	543	543	543	543	540	543
Fuel Used (Eng) - Gal	93.5	94.7	95.8	97.0	98.1	99.1	100.2	101.2	102.1
Fuel Used (A/B) - Gal	152.6	156.2	159.4	162.6	166.0	168.8	171.8	174.6	177.2
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.	0.1	0.1	0.1	0.1	0.1	0.1	0.1	0.1	0.1
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	O	O	O	O	O	O	O	O	O
Time - Sec	97.1	103	109	115	121	126.2	131.7	137	142.8

Test	CHECK CLIMB ~ MAX POWER								
Flight No.	20	20	20	20	20	20	20	20	20
Run No.									
Altitude - Ft	25065	26060	26980	27990	29015	29980	30995	31975	33010
IAS - Knots	373.0	366.0	361.5	356.5	351.5	346.0	337.0	334.0	327.0
OAT - °C	15.5	13.5	11.5	8.5	6.5	3.5	0	-2	-5.5
RPM (high press)	9470	9465	9465	9465	9445	9415	9390	9365	9340
RPM (low press)	5770	5785	5800	5820	5825	5820	5820	5800	5785
Exhaust Gas Temp - °C	543	545	550	553	553	553	550	550	545
Fuel Used (Eng) - Gal	103.2	104.3	105.1	106.2	107.3	108.2	109.2	110.2	111.2
Fuel Used (A/B) - Gal	180.2	183.2	185.6	188.4	191.4	194.0	196.6	199.2	202.0
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.	0.1	0.1	0.1	0.1	0.1	0.1	0.1	0	0
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	O	O	O	O	O	O	O	O	O
Time - Sec	148.7	154.5	160	166	172.7	178.8	179.4	191	198

Test	CHECK CLIMB ~ MAX POWER								
Flight No.	20	20	20	20	20	20	20	20	20
Run No.									
Altitude - Ft	33970	34955	35925	36930	37935	38915	39910	40930	41970
IAS - Knots	318.5	309.0	303.5	296.5	289.0	285.0	277.5	268.0	263.0
OAT - °C	-7.5	-10	-13.5	-16	-14.5	-16.5	-16.5	-18	-19.5
RPM (high press)	9315	9265	9230	9220	9190	9165	9165	9140	9110
RPM (low press)	5785	5760	5750	5760	5730	5716	5735	5735	5720
Exhaust Gas Temp - °C	545	545	543	543	543	540	543	545	545
Fuel Used (Eng) - Gal	112.0	112.9	114.0	115.1	116.0	117.1	118.1	119.3	120.4
Fuel Used (A/B) - Gal	204.2	207.0	210.0	213.0	216.0	219.0	222.0	225.2	228.8
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.	0	0	0	0	0	0	0.2	0.33	0.7
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	O	O	O	O	O	O	O	O	O
Time - Sec	204	211	220	228.8	237.9	247.4	258	269.1	281.9

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Test	CHECK CLIMB ~ MAX POWER								
Flight No.	20	20	20	20	20	20	20	20	20
Run No.									
Altitude - Ft	42870	42975	43495	43990	44490	45010	45485	46015	46480
IAS - Knots	262.0	262.0	261.5	257.5	253.5	248.5	245.5	242.0	238.5
OAT - °C	-20.5	-20.5	-21.5	-22	-23	-24	-24.8	-26	-27
RPM (high press)	9110	9110	9095	9095	9085	9080	9070	9060	9045
RPM (low press)	5725	5720	5720	5730	5735	5740	5740	5745	5735
Exhaust Gas Temp - °C	550	550	550	550	553	553	557	557	563
Fuel Used (Eng) - Gal	121.8	122.0	122.8	123.5	124.2	124.9	125.5	126.2	126.9
Fuel Used (A/B) - Gal	232.8	233.4	235.6	237.8	239.6	241.6	243.4	245.6	247.6
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.	1.4	1.93	2.3	2.6	3.05	3.33	3.50	4.40	5.10
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	O	O	O	O	O	O	O	O	O
Time - Sec	296.9	299	308	316.4	324	332.1	340	349	358

Test	CHECK CLIMB ~ MAX POWER								
Flight No.	20	20	20	20	20	20	20	20	20
Run No.									
Altitude - Ft	47020	47630	48040	48515	49005	49730	50265	50660	50725
IAS - Knots	231.0	227.0	227.5	228.0	226.5	221.5	210.5	207.0	209.0
OAT - °C	-28	-30	-30	-30	-32	-32	-34	-35	-34.5
RPM (high press)	9045	9020	8990	8990	8990	8990	8990	8990	8960
RPM (low press)	5765	5735	5780	5735	5755	5775	5785	5805	5820
Exhaust Gas Temp - °C	563	567	567	570	570	570	575	583	575
Fuel Used (Eng) - Gal	127.6	128.4	129.3	130.4	131.4	132.5	133.5	134.5	135.1
Fuel Used (A/B) - Gal	249.6	252.0	254.6	257.6	260.8	263.8	266.8	269.6	271.4
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.	5.75	6.80	7.85	8.40	9.00	10.70	11.75	13.70	15.95
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	O	O	O	O	O	O	O	O	O
Time - Sec	367.2	379	390.8	405.2	420.2	435	450	465	475

Test	CHECK CLIMB ~ MAX POWER								
Flight No.	21	21	21	21	21	21	21	21	21
Run No.									
Altitude - Ft	3090	4095	4985	5935	6975	7915	8960	9980	10950
IAS - Knots	485.25	485.0	477.0	474.5	462.0	455.5	449.0	448.0	446.80
OAT - °C	66	66	64	62	58.5	56	53	50	48
RPM (high press)	9480	9500	9505	9505	9495	9485	9480	9465	9480
RPM (low press)	5420	5440	5445	5460	5465	5480	5515	5515	5545
Exhaust Gas Temp - °C	485	485	490	510	515	515	520	520	520
Fuel Used (Eng) - Gal	52.4	54.7	56.1	57.5	58.8	60.1	61.7	63.4	65.0
Fuel Used (A/B) - Gal	95.6	102.8	107.4	111.8	116.2	120.2	125.2	130.6	134.8
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.	0	0	0	0	0	0	0	0	0
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	O	O	O	O	O	O	O	O	O
Time - Sec	0	10	17	23.5	30	36	44	52.5	60

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Test	CHECK CLIMB ~ MAX POWER								
Flight No.	21	21	21	21	21	21	21	21	21
Run No.									
Altitude - Ft	11965	12950	13970	14920	15930	16955	17935	18970	20005
IAS - Knots	445.85	442.0	437.5	431.0	424.5	416.75	409.5	404.75	397.25
OAT - °C	46.0	45.5	40.0	37.0	34.0	30.5	29.5	27.5	25.0
RPM (high press)	9480	9480	9465	9460	9460	9460	9470	9465	9460
RPM (low press)	5550	5560	5570	5580	5605	5635	5660	5670	5690
Exhaust Gas Temp - °C	522.5	522.5	530	530	540	540	540	540	540
Fuel Used (Eng) - Gal	65.8	68.2	69.8	70.9	72.1	73.2	74.2	75.1	76.3
Fuel Used (A/B) - Gal	140.6	145.2	149.8	153.4	156.6	159.8	162.6	165.6	168.8
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.	0	0	0	0	0	0	0	0	0
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	O	O	O	O	O	O	O	O	O
Time - Sec	68	76	83.5	89.5	95	100.5	105	111	117

Test	CHECK CLIMB ~ MAX POWER								
Flight No.	21	21	21	21	21	21	21	21	21
Run No.									
Altitude - Ft	21035	21980	22985	23990	25025	26050	27020	28020	29030
IAS - Knots	397.25	393.75	388.0	379.5	376.75	370.25	364.0	355.5	360.25
OAT - °C	24.0	22.0	20.0	18.0	16.5	14.0	12.0	9.0	6.0
RPM (high press)	9460	9460	9460	9460	9460	9460	9460	9460	9445
RPM (low press)	5690	5705	5725	5755	5770	5790	5800	5825	5835
Exhaust Gas Temp - °C	540	540	540	540	540	540	550	550	555
Fuel Used (Eng) - Gal	77.6	78.7	79.6	80.7	81.8	82.8	83.8	84.0	85.6
Fuel Used (A/B) - Gal	172.6	175.6	178.4	181.2	184.4	187.2	189.8	192.2	194.8
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.	0	0	0	0	0	0	0	0	0
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	O	O	O	O	O	O	O	O	O
Time - Sec	124	130	135	140.5	147	153	159	164	169.5

Test	CHECK CLIMB ~ MAX POWER								
Flight No.	21	21	21	21	21	21	21	21	21
Run No.									
Altitude - Ft	30010	31005	31990	33010	33955	34970	35940	36960	37910
IAS - Knots	343.25	333.75	327.0	323.0	318.5	314.75	303.5	300.0	286.5
OAT - °C	2.5	-0.5	-4.0	-6.0	-8.0	-10.0	-12.0	-14.0	-15.5
RPM (high press)	9440	9400	9365	9310	9335	9250	9235	9215	9210
RPM (low press)	5840	5835	5810	5790	5765	5760	5760	5745	5760
Exhaust Gas Temp - °C	555	555	555	542.5	572.5	540	540	540	540
Fuel Used (Eng) - Gal	86.5	87.3	88.3	89.5	90.3	91.8	92.7	93.7	94.5
Fuel Used (A/B) - Gal	197.0	199.4	202.0	205.2	208.0	211.6	214.4	217.0	219.4
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.	0	0	0	0	0	0	0	0	0
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	O	O	O	O	O	O	O	O	O
Time - Sec	175	180.5	187	195	203	212.5	220	228	236

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Test	CHECK CLIMB ~ MAX POWER								
Flight No.	21	21	21	21	21	21	21	21	21
Run No.									
Altitude - Ft	38930	39945	40930	41970	42860	43730	44710	45785	46715
IAS - Knots	281.5	271.5	266.5	259.5	256.0	255.0	248.0	237.25	225.7
OAT - °C	-16.0	-18.0	-20.	-21	-22	-22.5	-24	-26	-28
RPM (high press)	9190	9175	9110	9120	9095	9090	9065	9065	9070
RPM (low press)	5740	5745	5715	5735	5730	5730	5730	5745	5750
Exhaust Gas Temp - °C	540	542.5	542.5	555	555	555	555	560	565
Fuel Used (Eng) - Gal	95.5	96.4	97.4	98.6	99.9	101.1	102.5	103.6	104.8
Fuel Used (A/B) - Gal	222.2	224.6	227.8	231.2	235.0	238.8	242.6	246.0	249.4
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.	0	0	0	0.3	1.4	2.1	2.6	3.8	4.4
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	O	O	O	O	O	O	O	O	O
Time - Sec	245	255	264	277	292	307	322	337	352

Test	CHECK CLIMB ~ MAX POWER								
Flight No.	21	21	21	21	21	21	21	21	21
Run No.									
Altitude - Ft	47155	47465	47975	48840	49630	49980	50235	50255	50310
IAS - Knots	228.0	230.5	231.0	225.2	213.5	209.5	208.0	210.5	211.5
OAT - °C	-29	-29	-29	-30	-32	-34	-34	-34	-33
RPM (high press)	9010	9010	9010	9000	9010	8990	8970	8970	8970
RPM (low press)	5735	5730	5740	5745	5790	5765	5745	5740	5735
Exhaust Gas Temp - °C	560	565	565	582.5	582.5	582.5	582.5	582.5	582.5
Fuel Used (Eng) - Gal	105.9	106.9	108.1	109.2	110.3	111.2	112.2	113.2	114.2
Fuel Used (A/B) - Gal	252.6	255.8	259.0	262.2	265.2	268.2	271.3	274.0	276.8
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.	6.3	7.7	8.4	9.3	10.7	12.2	14.2	15.75	17.9
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	O	O	O	O	O	O	O	O	O
Time - Sec	367	382	397	412	427	442	457	472	487

Test	CHECK CLIMB ~ MAX POWER								
Flight No.	21	21	21	21	21	21	21	21	21
Run No.									
Altitude - Ft	50380	50340	50455	50405	50550	50590	50655	50710	50695
IAS - Knots	210.5	213.5	212.5	214.0	210.0	206.5	201.5	202.5	201.0
OAT - °C	-32.5	-32	-32	-32	-33	-34	-35	-35	-34
RPM (high press)	8970	8970	8970	8970	8970	8970	8940	8940	8935
RPM (low press)	5735	5730	5725	5745	5750	5770	5730	5725	5715
Exhaust Gas Temp - °C	582.5	582.5	575	575	575	582.5	582.5	582.5	582.5
Fuel Used (Eng) - Gal	115.0	116.2	117.2	118.1	119.0	120.0	120.9	121.9	122.0
Fuel Used (A/B) - Gal	279.6	282.5	285.6	288.4	291.2	294.0	296.8	299.6	302.4
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.	18.4	18.4	18.4	18.4	18.4	18.4	18.4	18.4	18.4
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	O	O	O	O	O	O	O	O	O
Time - Sec	502	517	532	547	562	577	592	607	622

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Test	CHECK CLIMB ~ MAX POWER								
Flight No.	21	21	21	21	21	21	21	21	21
Run No.									
Altitude - Ft	50550	50240	50765	51100	51750	52280	52730	53295	53775
IAS - Knots	207.5	218.0	209.5	207.0	202.5	197.5	195.25	192.15	186.0
OAT - °C	-34	-32	-32	-32.5	-34	-36	-37	-38	-40
RPM (high press)	8935	8925	8940	8940	8940	8925	8910	8865	8890
RPM (low press)	5710	5695	5715	5720	5715	5725	5725	5755	5775
Exhaust Gas Temp - °C	582.5	582.5	582.5	582.5	572.5	582.5	582.5	585	585
Fuel Used (Eng) - Gal	123.8	124.8	126.6	128.0	129.9	132.7	137.3	146.0	150.8
Fuel Used (A/B) - Gal	305.2	308.2	313.6	317.8	323.4	331.6	345.4	371.8	386.2
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.	18.4	18.4	18.4	18.4	18.4	18.4	18.4	18.4	18.4
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	O	O	O	O	O	O	O	O	O
Time - Sec	637	652	687	704	724	780	799	949	1032

Test	CHECK CLIMB ~ MAX POWER								
Flight No.	21	21							
Run No.									
Altitude - Ft	54390	54735							
IAS - Knots	177.5	173.0							
OAT - °C	-42	-42							
RPM (high press)	8840	8765							
RPM (low press)	585	585							
Exhaust Gas Temp - °C	585	585							
Fuel Used (Eng) - Gal	153.1	154.2							
Fuel Used (A/B) - Gal	393.2	396.6							
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.	18.4	18.4							
Bleed Valve Position	C	C							
Fuel Weight - lbs/gal									
Nozzle Position	O	O							
Time - Sec	1073	1095							

Test									
Flight No.									
Run No.									
Altitude - Ft									
IAS - Knots									
OAT - °C									
RPM (high press)									
RPM (low press)									
Exhaust Gas Temp - °C									
Fuel Used (Eng) - Gal									
Fuel Used (A/B) - Gal									
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.									
Bleed Valve Position									
Fuel Weight - lbs/gal									
Nozzle Position									
Time - Sec									

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Test	CHECK CLIMB – MILITARY POWER								
Flight No.	1	1	1	1	1	1	1	1	1
Run No.									
Altitude – Ft	3999	4957	5974	6977	7991	8987	9989	10978	11978
IAS – Knots	404.3	401.8	401.8	397.3	391.3	396.3	384.9	382.9	376.7
OAT – °C	56.0	54.0	52.0	48.0	46.0	43.0	39.0	36.0	36.0
RPM (high press)	9545	9540	9545	9545	9545	9545	9545	9545	9555
RPM (low press)	5330	5550	5660	5580	5600	5640	5655	5670	5660
Exhaust Gas Temp – °C	523	523	527	527	525	540	540	540	543
Fuel Used (Eng) – Gal	72.0	79.0	89.0	96.0	105	111	115	121	126
Fuel Used (A/B) – Gal	76.6	76.6	76.6	76.6	76.6	76.6	76.6	76.6	76.6
Fuel Flow (total) – GPH									
Oil Cooler GAP – in.°C	0.1	0.1	0.1	0.1	0.1	0.1	0.35	0.35	0.35
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight – lbs/gal									
Nozzle Position	C	C	C	C	C	C	C	C	C
Time – Sec	0	30.5	74.6	102.6	146.6	177.5	195.4	223.6	246.6

Test	CHECK CLIMB – MILITARY POWER								
Flight No.	1	1	1	1	1	1	1	1	1
Run No.									
Altitude – Ft	12968	13990	14970	15989	16993	17998	18972	19976	20975
IAS – Knots	364.2	378.3	366.2	367.7	360.7	357.7	354.2	350.7	346.2
OAT – °C	34.0	36.0	32.5	31.5	28.5	26.0	23.5	21.5	19.5
RPM (high press)	9555	9545	9565	9540	9555	9545	9540	9480	9545
RPM (low press)	5680	5670	5710	5720	5740	5765	5790	5780	5840
Exhaust Gas Temp – °C	543	546	543	540	540	540	543	519	540
Fuel Used (Eng) – Gal	130	140	143	149	152	157	161	183	187
Fuel Used (A/B) – Gal	76.6	76.6	76.6	76.6	76.6	76.6	76.6	76.6	76.6
Fuel Flow (total) – GPH									
Oil Cooler GAP – in.°C	0.55	1.55	1.80	2.2	2.2	2.2	2.5	2.3	8.4
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight – lbs/gal									
Nozzle Position	C	C	C	C	C	C	C	C	C
Time – Sec	267.4	311.9	326.2	358.8	376.2	398.6	420.6	566.8	589.6

Test	CHECK CLIMB – MILITARY POWER								
Flight No.	1	1	1	1	1	1	1	1	1
Run No.									
Altitude – Ft	21970	22980	23954	24957	25944	26929	27925	28896	29850
IAS – Knots	341.2	338.8	334.3	332.3	328.9	324.0	320.0	320.5	313.3
OAT – °C	16.5	14.5	12.0	10.0	6.5	4.0	1.0	0	-4.0
RPM (high press)	9545	9530	9510	9475	9455	9410	9380	9350	9325
RPM (low press)	5855	5870	5870	5840	5840	5820	5810	5800	5795
Exhaust Gas Temp – °C	546	550	546	543	543	590	590	590	525
Fuel Used (Eng) – Gal	192	197	200	205	209	212	216	222	[illegible]
Fuel Used (A/B) – Gal	76.6	76.6	76.6	76.6	76.6	76.6	76.6	76.6	76.6
Fuel Flow (total) – GPH									
Oil Cooler GAP – in. °C	8.4	8.4	7.45	7.45	7.45	2.45	7.45	7.45	7.45
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight – lbs/gal									
Nozzle Position	C	C	C	C	C	C	C	C	C
Time – Sec	614.9	639.7	658.9	682.9	707.5	728.6	752.5	788.1	809.9

DATA CORRECTED FOR INSTRUMENT ERROR
YF-100 USAF No. 52-5754

Test	CHECK CLIMB ~ MILITARY POWER								
Flight No.	1	1	1	1	1	1	1	1	1
Run No.									
Altitude - Ft	30818	31783	32760	33729	34696	35750	36603	37587	38523
IAS - Knots	308.4	306.5	302.0	297.0	292.2	285.6	281.8	276.7	269.5
OAT - °C	-6.0	-8.5	-12.0	-14.0	-15.5	-18.0	-19.5	-20.0	-21.5
RPM (high press)	9280	9260	9205	9200	9175	9150	9130	9100	9095
RPM (low press)	5780	5770	5745	5750	5755	5740	5745	5755	5750
Exhaust Gas Temp - °C	535	531	519	519	519	519	519	523	527
Fuel Used (Eng) - Gal	228	234	238	242	248	251	258	264	270
Fuel Used (A/B) - Gal	76.8	76.8	77.0	77.0	77.2	77.4	77.6	77.8	78.0
Fuel Flow (total) - GPH									
Oil Cooler GAP - in. °C	7.45	7.45	7.45	7.45	7.7	7.7	8.8	10.2	11.6
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	C	C	C	C	C	C	C	C	C
Time - Sec	834.9	873.6	910.8	938.1	982.9	1014.9	1070.4	1128.6	1184.9

Test	CHECK CLIMB ~ MILITARY POWER								
Flight No.	1	1	1	1		7	7	7	7
Run No.									
Altitude - Ft	39515	40493	41475	42937		3050	4065	4360	5175
IAS - Knots	266.0	263.0	256.0	249.0		407.5	405.5	398.75	395.5
OAT - °C	-22.5	-23.5	-25.0	-27.5					
RPM (high press)	9085	9040	9035	8990		9530	9535	9535	9535
RPM (low press)	5730	5740	5745	5740		5510	5535	5545	5555
Exhaust Gas Temp - °C	535	535	540	535					
Fuel Used (Eng) - Gal	278	307	313	335		59.5	70.2	72.8	80.3
Fuel Used (A/B) - Gal	78.4	78.6	78.6	78.6		66.8	66.8	66.8	66.8
Fuel Flow (total) - GPH									
Oil Cooler GAP - in. °C	14.0	18.4	18.4	18.4		0	0	0	0
Bleed Valve Position	C	C	C	C		C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	C	C	C	C		C	C	C	C
Time - Sec	1266.6	1568.4	1634.8	1875.9		0	48.5	60.5	95.5

Test	CHECK CLIMB ~ MILITARY POWER								
Flight No.	7	7	7	7	7	7	7	7	7
Run No.									
Altitude - Ft	5330	6030	6535	7025	8015	8590	8975	9350	9915
IAS - Knots	400.5	394.5	394.0	394.5	391.0	387.5	390.0	385.0	382.5
OAT - °C									
RPM (high press)	9535	9535	9535	9535	9535	9535	9535	9535	9535
RPM (low press)	5555	5575	5585	5590	5610	5615	5625	5645	5665
Exhaust Gas Temp - °C									
Fuel Used (Eng) - Gal	83.3	87.5	91.5	95.9	101.7	104.7	108.0	110.7	114.5
Fuel Used (A/B) - Gal	66.8	66.8	66.8	66.8	66.8	66.8	66.8	66.8	66.8
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.	0	0	0	0	0	0	0	0	0
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	C	C	C	C	C	C	C	C	C
Time - Sec	114	128	146.5	167.5	194.5	207.5	223	236	253.5

DATA CORRECTED FOR INSTRUMENT ERROR
YF-100 USAF No. 52-5754

Test	CHECK CLIMB - MILITARY POWER								
Flight No.	7	7	7	7	7	7	7	7	7
Run No.									
Altitude - Ft	10770	11690	12740	13905	14300	15100	15935	16855	17605
IAS - Knots	381.0	379.0	376.0	368.5	365.5	366.0	363.0	360.5	359.0
OAT - °C									
RPM (high press)	9526	9525	9525	9525	9525	9510	9510	9510	9510
RPM (low press)	5660	5685	5715	5725	5740	5770	5785	5795	5800
Exhaust Gas Temp - °C									
Fuel Used (Eng) - Gal	119.7	125.2	131.3	135.8	137.9	141.6	145.4	149.7	152.3
Fuel Used (A/B) - Gal	66.8	66.8	66.8	66.8	66.8	66.8	66.8	66.8	66.8
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.	0.3	0.3	0.65	0.55	0.55	0.55	0.55	0.55	0.75
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	C	C	C	C	C	C	C	C	C
Time - Sec	278	303.5	333.5	354	364.5	382.4	401	421.5	435.5

Test	CHECK CLIMB - MILITARY POWER								
Flight No.	7	7	7	7	7	7	7	7	7
Run No.									
Altitude - Ft	18440	18985	19455	19965	20700	21005	21995	22840	23275
IAS - Knots	354.5	354.0	352.0	349.0	343.5	344.0	344.5	338.5	335.5
OAT - °C									
RPM (high press)	9510	9515	9515	9515	9515	9505	9500	9485	9460
RPM (low press)	5800	5815	5815	5840	5855	5855	5855	5845	5845
Exhaust Gas Temp - °C									
Fuel Used (Eng) - Gal	156.2	158.1	159.5	161.3	164.7	166.4	169.9	172.5	174.6
Fuel Used (A/B) - Gal	66.8	66.8	66.8	66.8	66.8	66.8	66.8	67.0	67.0
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.	0.75	1.0	1.0	1.0	1.0	1.25	1.25	1.25	1.5
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	C	C	C	C	C	C	C	C	C
Time - Sec	454.5	464.5	471.5	480.5	493	507.5	526.5	541	552

Test	CHECK CLIMB - MILITARY POWER								
Flight No.	7	7	7	7	7	7	7	7	7
Run No.									
Altitude - Ft	24025	24970	25300	26035	26455	27020	27470	27950	28855
IAS - Knots	336.5	331	328.5	327	326.5	325.5	323.5	321.5	317
OAT - °C									
RPM (high press)	9450	9435	9415	9395	9390	9360	9350	9340	9315
RPM (low press)	5835	5830	5830	5820	5815	5805	5800	5795	5785
Exhaust Gas Temp - °C									
Fuel Used (Eng) - Gal	177.4	180.5	181.7	184.7	186.5	189.1	191.1	193.3	197
Fuel Used (A/B) - Gal	67.	67	67	67	67	67	67	67.2	67.2
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.	2.05	1.9	2.3	2.6	2.6	2.9	3.3	3.3	3.8
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	C	C	C	C	C	C	C	C	C
Time - Sec	568.5	586.5	593.5	611.5	623.5	639.5	652.5	671.5	682

DATA CORRECTED FOR INSTRUMENT ERROR
YF-100 USAF No. 52-5754

Test	CHECK CLIMB ~ MILITARY POWER								
Flight No.	7	7	7	7	7	7	7	7	7
Run No.									
Altitude - Ft	29505	29960	30685	30965	31335	31550	31820	32310	32975
IAS - Knots	313.5	313.0	309	306.5	304.5	307.5	305	307.5	307
OAT - °C									
RPM (high press)	9295	9275	9265	9250	9240	9230	9220	9220	9225
RPM (low press)	5765	5760	5760	5750	5735	5745	5715	5740	5750
Exhaust Gas Temp - °C									
Fuel Used (Eng) - Gal	199.4	201.5	203.9	204.8	207.1	208.9	205.1	213.9	215.9
Fuel Used (A/B) - Gal	67.2	67.4	67.4	67.4	67.4	67.4	67.8	68	68.2
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.	3.9	4.4	4.75	4.75	5.5	5.7	5.7	6.6	6.8
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	C	C	C	C	C	C	C	C	C
Time - Sec	708.5	723.5	741.5	749.5	765	778.5	791	818	837.5

Test	CHECK CLIMB ~ MILITARY POWER								
Flight No.	7	7	7	7	7	7	7	7	7
Run No.									
Altitude - Ft	33555	34085	34465	35105	35310	35765	36295	36730	37050
IAS - Knots	297.5	296.5	294.5	287	289	287	281.5	277.5	275.5
OAT - °C									
RPM (high press)	9215	9190	9190	9170	9155	9145	9140	9120	9115
RPM (low press)	5735	5730	5745	5740	5725	5720	5735	5725	5725
Exhaust Gas Temp - °C									
Fuel Used (Eng) - Gal	218.1	220.2	221.3	224	225.8	227.8	229.8	232.1	233.9
Fuel Used (A/B) - Gal	68.4	68.1	68.6	68.8	69	69	69.4	69.6	69.8
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.	6.9	6.9	7.1	7.7	8.0	8.35	8.6	9.0	9.75
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	C	C	C	C	C	C	C	C	C
Time - Sec	852	869.5	851.5	903	919.5	937.5	956.5	978	996

Test	CHECK CLIMB ~ MILITARY POWER								
Flight No.	7	7	7	7	7	7	7	7	7
Run No.									
Altitude - Ft	37068	37675	38100	38235	38520	38670	39020	39205	39465
IAS - Knots	275	278.5	274.5	273.5	273	273	270.5	270	268.5
OAT - °C									
RPM (high press)	9100	9095	9095	9095	9070	9070	9065	9040	9040
RPM (low press)	5730	5705	5725	5735	5725	5730	5730	5725	5730
Exhaust Gas Temp - °C									
Fuel Used (Eng) - Gal	234	240.3	241.9	242.7	244.4	246	248.5	249.7	251.9
Fuel Used (A/B) - Gal	69.8	70	70	70.2	70.2	70.2	70.4	70.4	70.4
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.	9.75	11.4	11.4	11.7	12.5	12.5	13.	13.4	14
Bleed Valve Position	C	C	C	C	C	C	C	C	
Fuel Weight - lbs/gal									
Nozzle Position	C	C	C	C	C	C	C	C	C
Time - Sec	996.5	1058.5	1074.5	1082	1099.5	1117.5	1141.5	1154.5	1177.5

DATA CORRECTED FOR INSTRUMENT ERROR
YF-100 USAF No. 52-5754

Test	CHECK CLIMB ~ MILITARY POWER								
Flight No.	7	7	7	7	7	7	7	7	7
Run No.									
Altitude - Ft	39660	40060	40215	40420	40840	40855	40845	40975	40865
IAS - Knots	266.5	263	260	259	256.5	257	257.5	253.5	254
OAT - °C									
RPM (high press)	9040	9020	9020	9015	9020	9020	8995	8995	8995
RPM (low press)	5725	5725	5725	5720	5725	5725	5705	5730	5720
Exhaust Gas Temp - °C									
Fuel Used (Eng) - Gal	253.5	256.9	258.2	259.5	264.5	265	269.9	271.3	278.5
Fuel Used (A/B) - Gal	70.4	70.4	70.4	70.4	70.4	70.4	70.4	70.4	70.4
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.	14.3	15.7	16.2	16.7	18.4	18.4	18.4	18.4	18.4
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	C	C	C	C	C	C	C	C	C
Time - Sec	1194	1231	1245.5	1259.5	1315.5	1322	1377	1393.5	1474.5

Test	CHECK CLIMB ~ MILITARY POWER								
Flight No.	7	7	7	7					
Run No.									
Altitude - Ft	41360	41510	41655	41890					
IAS - Knots	247.5	246	245.5	243					
OAT - °C									
RPM (high press)	8995	8960	8960	8970					
RPM (low press)	5725	5720	5720	5715					
Exhaust Gas Temp - °C									
Fuel Used (Eng) - Gal	280.8	282.3	283.7	286.2					
Fuel Used (A/B) - Gal	70.4	70.4	70.4	70.4					
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.	18.4	18.4	18.4	18.4					
Bleed Valve Position	C	C	C	C					
Fuel Weight - lbs/gal									
Nozzle Position	C	C	C	C					
Time - Sec	1501	1518.5	1535	1564.5					

Test									
Flight No.									
Run No.									
Altitude - Ft									
IAS - Knots									
OAT - °C									
RPM (high press)									
RPM (low press)									
Exhaust Gas Temp - °C									
Fuel Used (Eng) - Gal									
Fuel Used (A/B) - Gal									
Fuel Flow (total) - GPH									
Oil Cooler GAP - in									
Bleed Valve Position									
Fuel Weight - lbs/gal									
Nozzle Position									
Time - Sec									

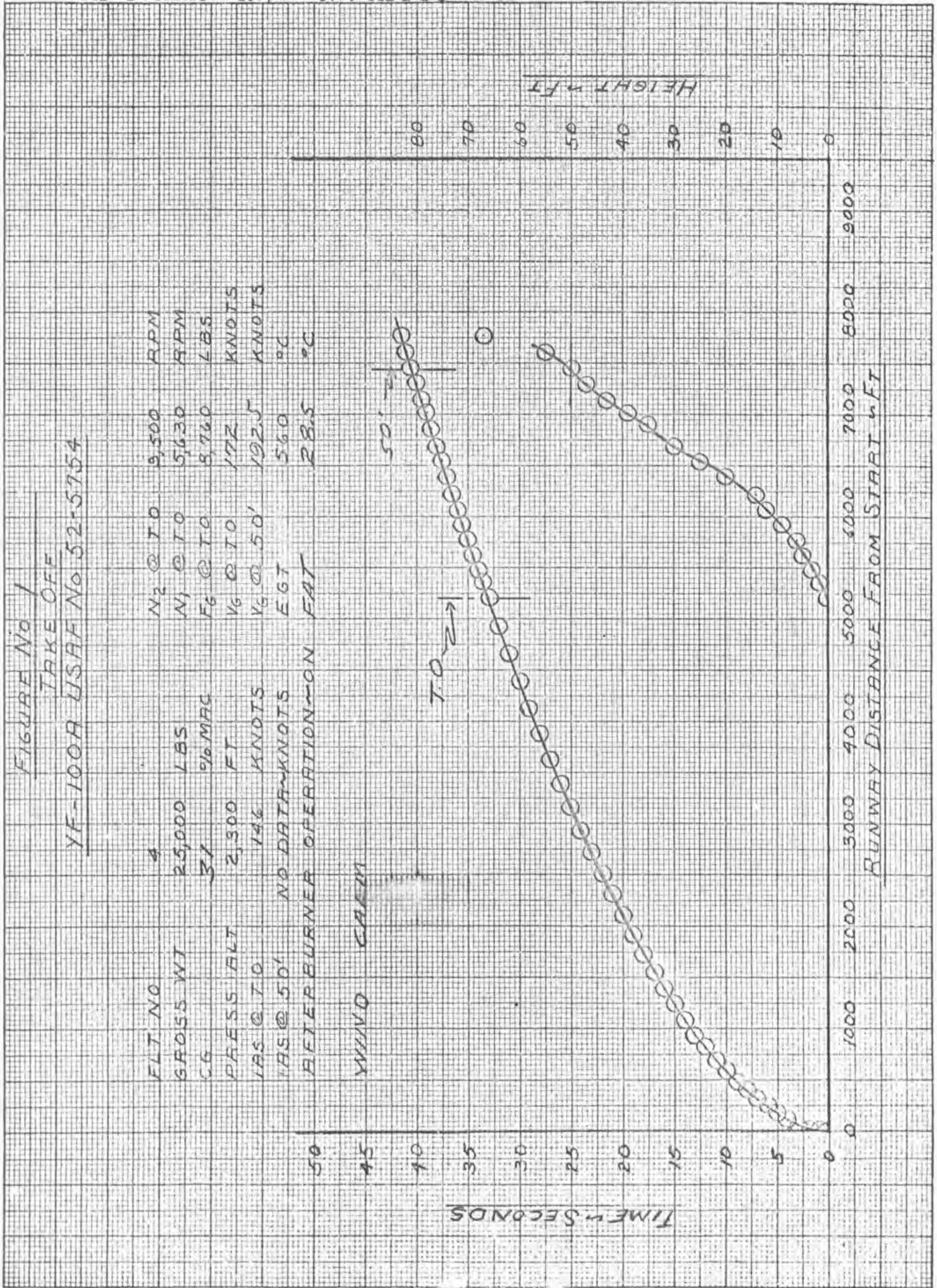
FIGURE No 1
TAKE OFF
YF-100A USAF No 52-5754
FLT NO 4
GROSS WT 25,000 LBS
CG 31 % MAC
PRESS ALT 2,300 FT
IAS @ T.O 146 KNOTS
IAS @ 50' NO DATA ~ KNOTS
AFTERBURNER OPERATION ~ ON
WIND CALM
N2 @ T.O 9,500 RPM
N1 @ T.O 5630 RPM
FG @ T.O 8,760 LBS
VG @ T.O 172 KNOTS
VG @ 50' 192.5 KNOTS
EGT 560 °C
FAT 28.5 °C
T.O
50'
TIME ~ SECONDS
HEIGHT ~ FT
RUNWAY DISTANCE FROM START ~ FT

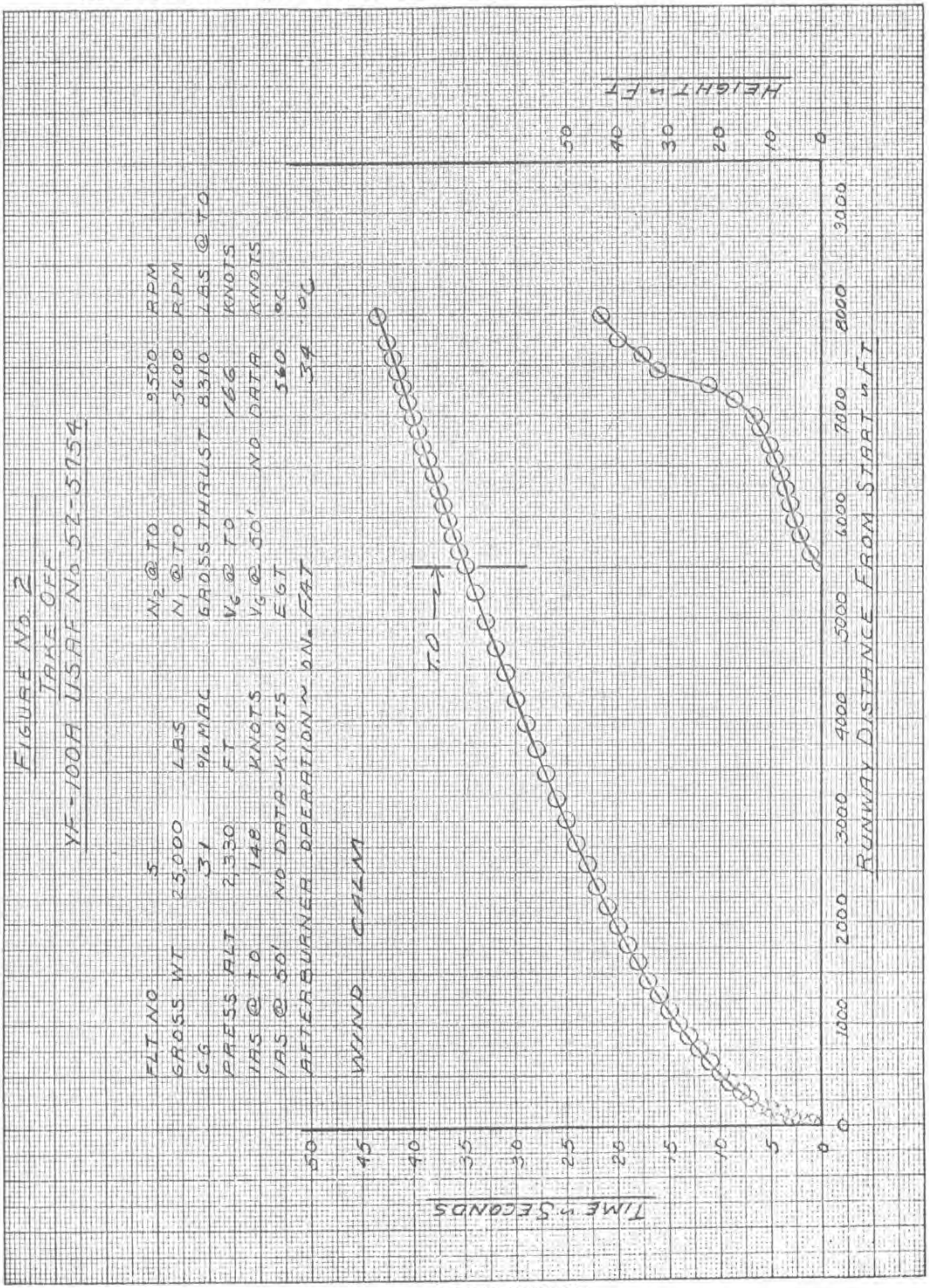
FIGURE No. 2
TAKE OFF
YF-100A USAF No 52-5754
FLT NO 5
GROSS WT 25,000 LBS
CG 31 % MAC
PRESS ALT 2,330 FT
IAS @ TO 148 KNOTS
IAS @ 50' NO DATA~KNOTS
AFTERBURNER OPERATION ~ ON. FAT 34 °C
WIND CALM
N2 @ TO 9500 RPM
N1 @ TO 5600 RPM
GROSS THRUST 8310 LBS @ TO
VG @ TO 166 KNOTS
VG @ 50' NO DATA KNOTS
EGT 560 °C
T.O
TIME ~ SECONDS
HEIGHT ~ FT
RUNWAY DISTANCE FROM START ~ FT
0 1000 2000 3000 4000 5000 6000 7000 8000 9000
0 5 10 15 20 25 30 35 40 45 50
0 10 20 30 40 50

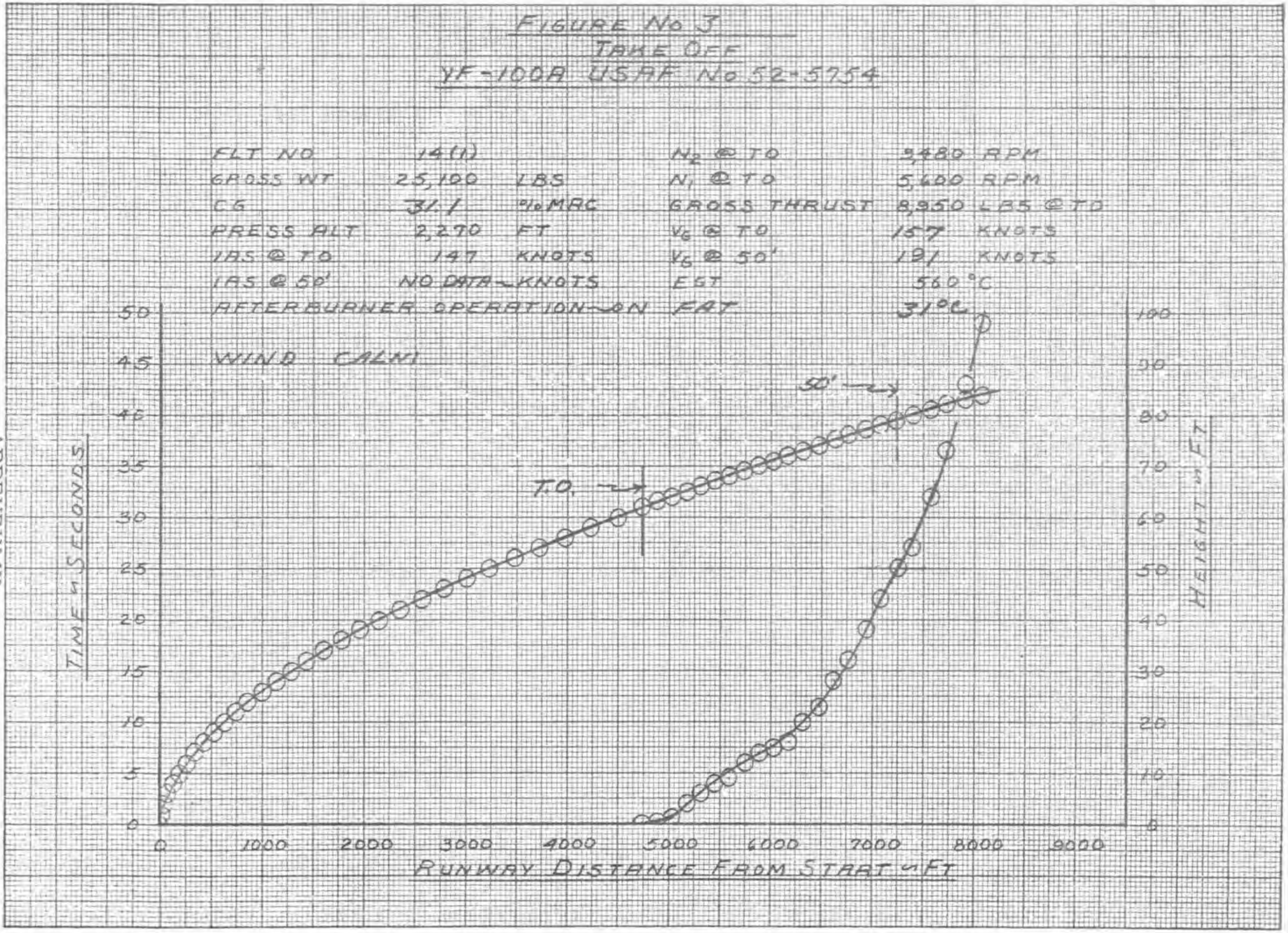
FIGURE No 3
TAKE OFF
YF-100A USAF No 52-5754
FLT NO 14(1)
GROSS WT 25,100 LBS
CG 31.1 % MAC
PRESS ALT 2,270 FT
IAS @ TO 147 KNOTS
IAS @ 50' NO DATA ~ KNOTS
AFTERBURNER OPERATION ~ ON
WIND CALM
N2 @ TO 3,480 RPM
N1 @ TO 5,600 RPM
GROSS THRUST 8,950 LBS @ TO
VG @ TO 157 KNOTS
VG @ 50' 191 KNOTS
EGT 560 °C
FAT 31°C
T.O.
50'
TIME ~ SECONDS
HEIGHT ~ FT
RUNWAY DISTANCE FROM START ~ FT
0
5
10
15
20
25
30
35
40
45
50
0
10
20
30
40
50
60
70
80
90
100
0
1000
2000
3000
4000
5000
6000
7000
8000
9000

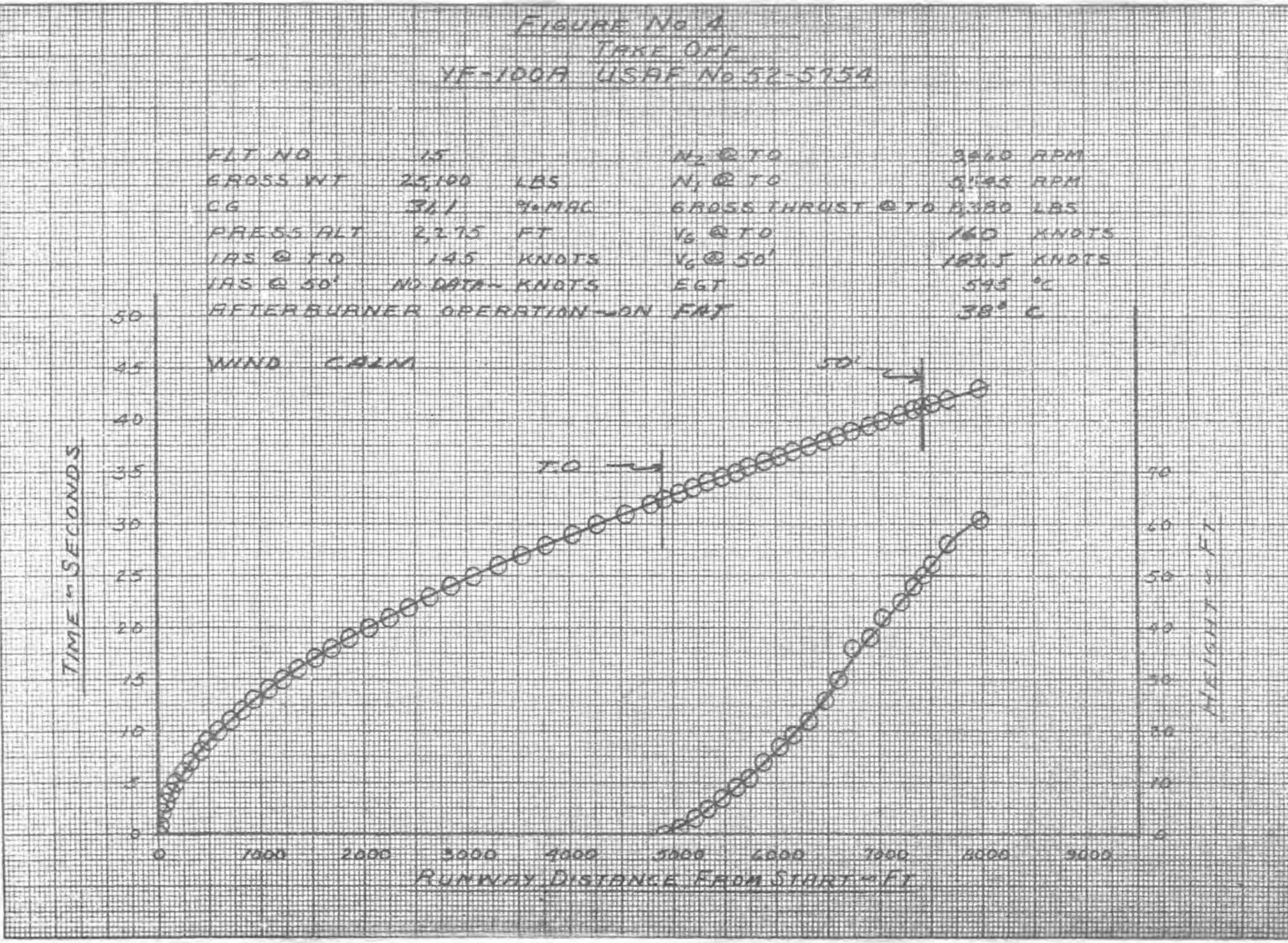
FIGURE No 4
TAKE OFF
YF-100A USAF No 52-5754
FLT NO 15
GROSS WT 25,100 LBS
CG 31.1 % MAC
PRESS ALT 2,275 FT
IAS @ TO 145 KNOTS
IAS @ 50' NO DATA ~ KNOTS
AFTERBURNER OPERATION ~ ON
WIND CALM
N_2 @ TO RPM
N_1 @ TO RPM
GROSS THRUST @ TO LBS
V_G @ TO 160 KNOTS
V_G @ 50' 183.5 KNOTS
EGT 595 °C
FAT 38° C
TO
50'
TIME ~ SECONDS
HEIGHT ~ FT
RUNWAY DISTANCE FROM START ~ FT
0
5
10
15
20
25
30
35
40
45
50
60
70
1000
2000
3000
4000
5000
6000
7000
8000
9000

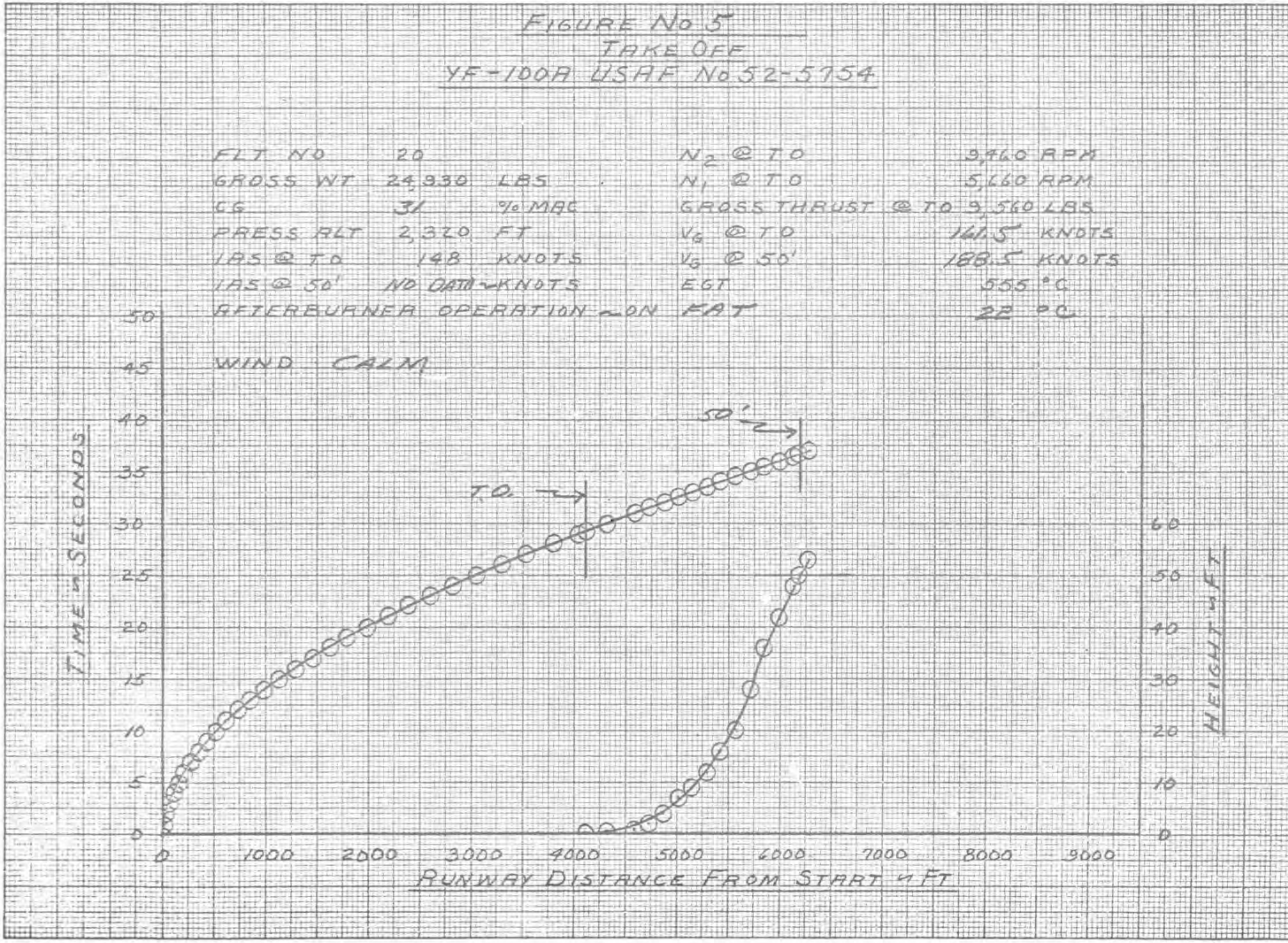

FIGURE No 5
TAKE OFF
YF-100A USAF No 52-5754

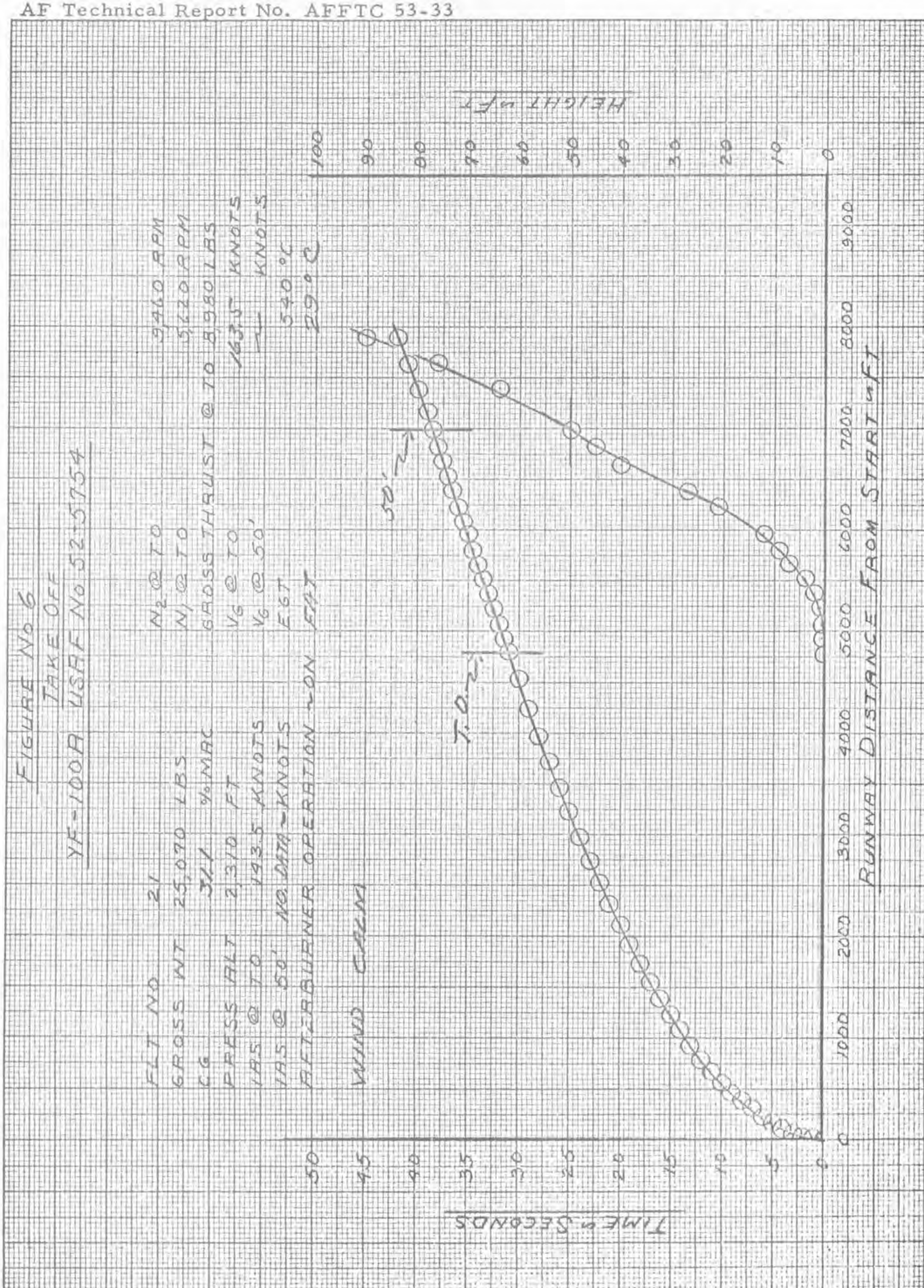
FIGURE No 6
TAKE OFF
YF-100A USAF No 52-5754
FLT NO 21
GROSS WT 25,070 LBS
C G 31.1 % MAC
PRESS ALT 2310 FT
IAS @ TO 143.5 KNOTS
IAS @ 50' NO DATA ~ KNOTS
AFTERBURNER OPERATION ~ ON
WIND CALM
N2 @ TO 9460 RPM
N1 @ TO 5620 RPM
GROSS THRUST @ TO 8980 LBS
VG @ TO 163.5 KNOTS
VG @ 50' — KNOTS
EGT 590°C
FAT 29°C
T.O.
50'
HEIGHT ~ FT
TIME ~ SECONDS
RUNWAY DISTANCE FROM START ~ FT
0
1000
2000
3000
4000
5000
6000
7000
8000
9000
0
5
10
15
20
25
30
35
40
45
50
0
10
20
30
40
50
60
70
80
90
100

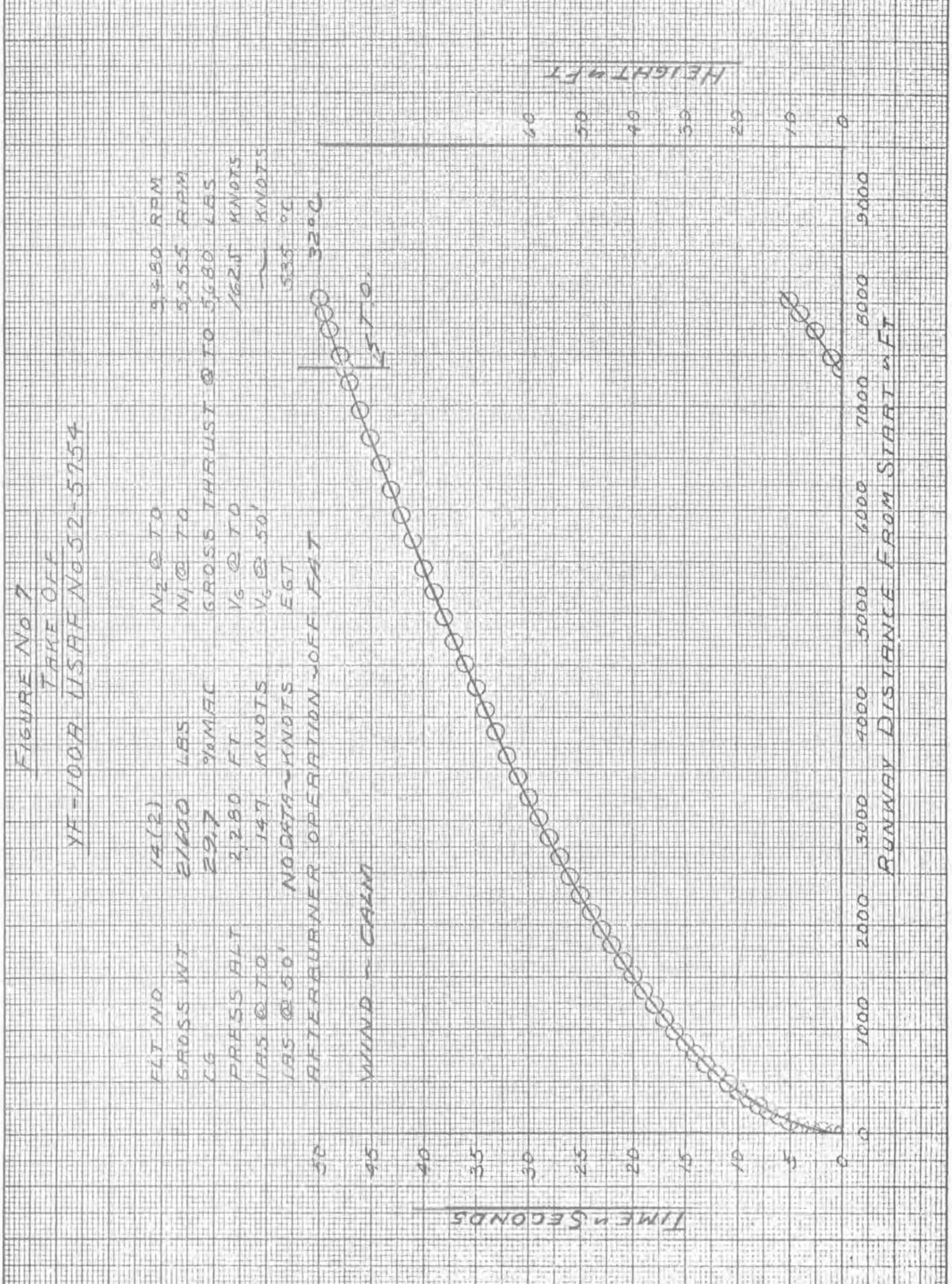
FIGURE No 7
TAKE OFF
YF-100A USAF No 52-5754
FLT NO 14(2)
GROSS WT 21600 LBS
CG 29.7 %MAC
PRESS ALT 2280 FT
IAS @ TO 147 KNOTS
IAS @ 50' NO DATA ~ KNOTS
AFTERBURNER OPERATION ~ OFF
WIND ~ CALM
N2 @ TO 9480 RPM
N1 @ TO 5555 RPM
GROSS THRUST @ TO 5680 LBS
VG @ TO 162.5 KNOTS
VG @ 50' ~ KNOTS
EGT 535 °C
FAT 32°C
T.O.
TIME ~ SECONDS
HEIGHT ~ FT
RUNWAY DISTANCE FROM START ~ FT

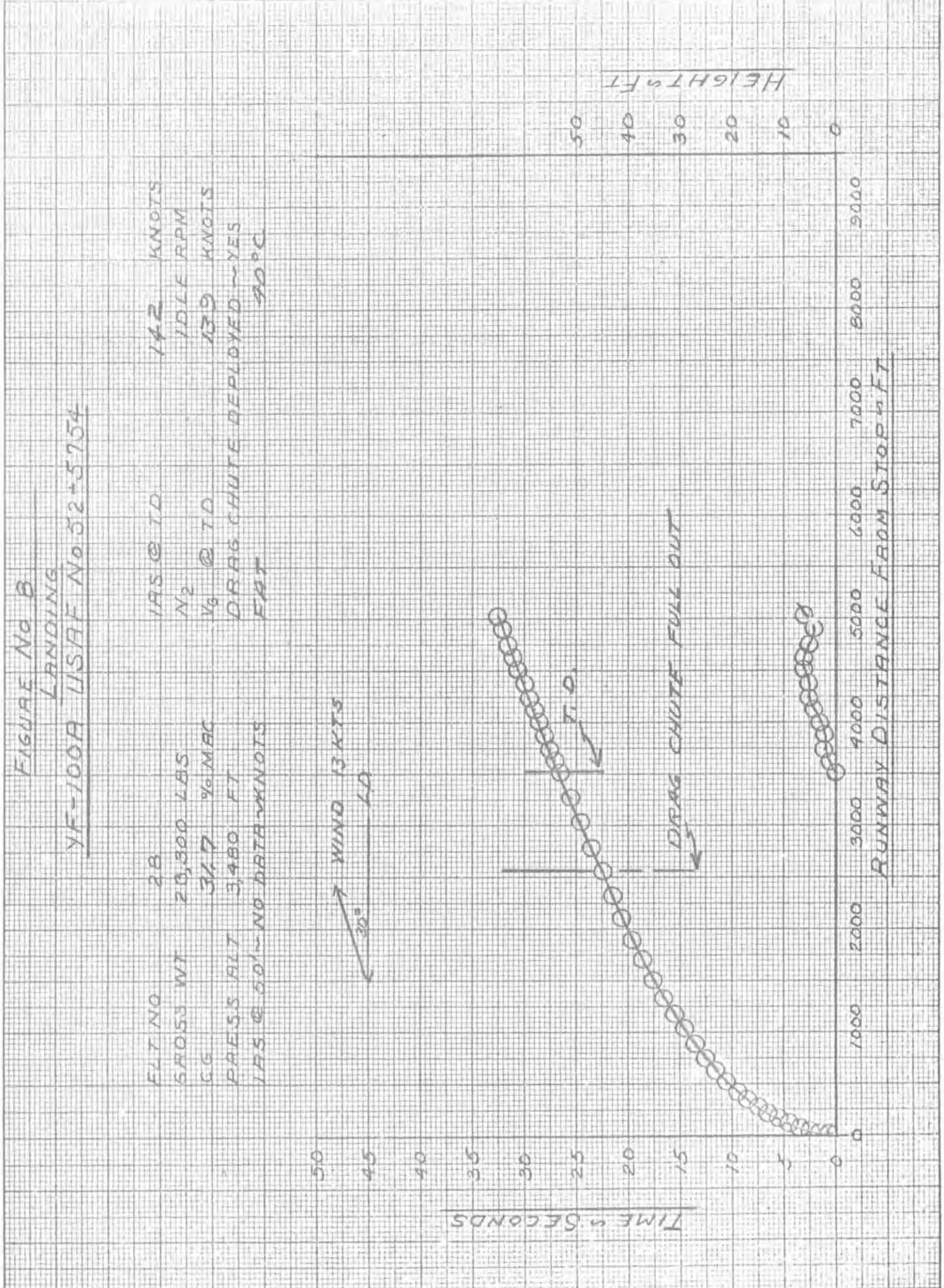
FIGURE No 8
LANDING
YF-100A USAF No 52-5754
FLT NO 28
GROSS WT 28,300 LBS
C G 31.7 % MAC
PRESS ALT 3,480 FT
IAS @ 50' ~ NO DATA ~ KNOTS
IAS @ TD 142 KNOTS
N_2 IDLE RPM
V_g @ TD 139 KNOTS
DRAG CHUTE DEPLOYED ~ YES
FAT
WIND 13 KTS
30°
T.D.
DRAG CHUTE FULL OUT
DRAG
HEIGHT ~ FT
0
10
20
30
40
50
TIME ~ SECONDS
0
5
10
15
20
25
30
35
40
45
50
RUNWAY DISTANCE FROM STOP ~ FT
0
1000
2000
3000
4000
5000
6000
7000
8000
9000

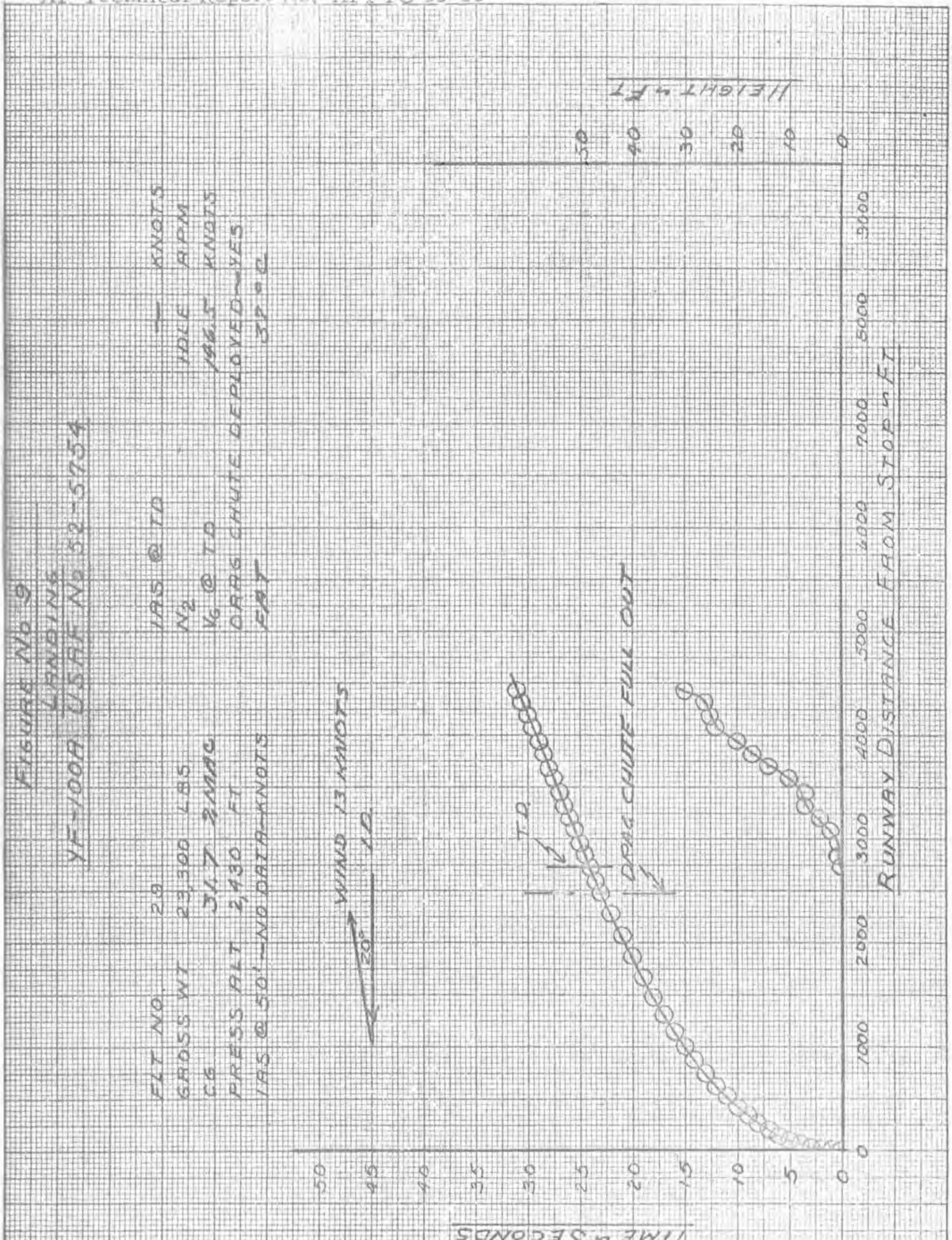
FIGURE No 9
LANDING
YF-100A USAF No 52-5754
FLT NO. 29
GROSS WT 23,300 LBS
CG 31.7 %MAC
PRESS ALT 2,430 FT
IAS @ 50' — NO DATA — KNOTS
IAS @ TD — KNOTS
N2 IDLE RPM
VG @ TD 146.5 KNOTS
DRAG CHUTE DEPLOYED — YES
FAT 37 °C
WIND 13 KNOTS
20°
L.D.
T.D.
DRAG CHUTE FULL OUT
TIME ~ SECONDS
HEIGHT ~ FT
RUNWAY DISTANCE FROM STOP ~ FT

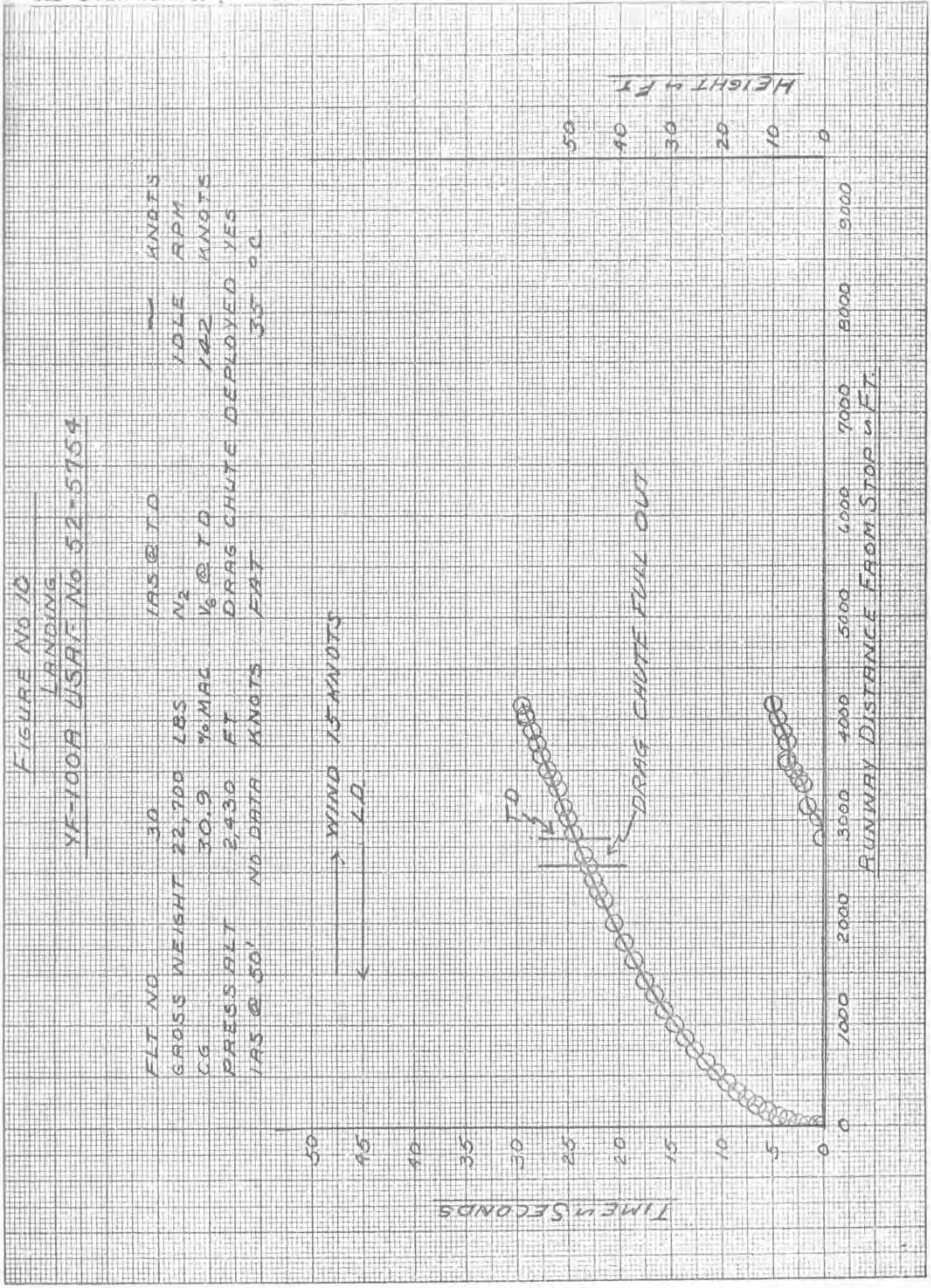
FIGURE No 10
LANDING
YF-100A USAF No 52-5754
FLT NO 30
GROSS WEIGHT 22,700 LBS
CG 30.9 % MAC
PRESS ALT 2,430 FT
IAS @ 50' NO DATA KNOTS
IAS @ TD ~ KNOTS
N2 IDLE RPM
VG @ TD 142 KNOTS
DRAG CHUTE DEPLOYED YES
FAT 35 °C
WIND 15 KNOTS
LD
TD
DRAG CHUTE FULL OUT
RUNWAY DISTANCE FROM STOP ~ FT
TIME ~ SECONDS
HEIGHT ~ FT

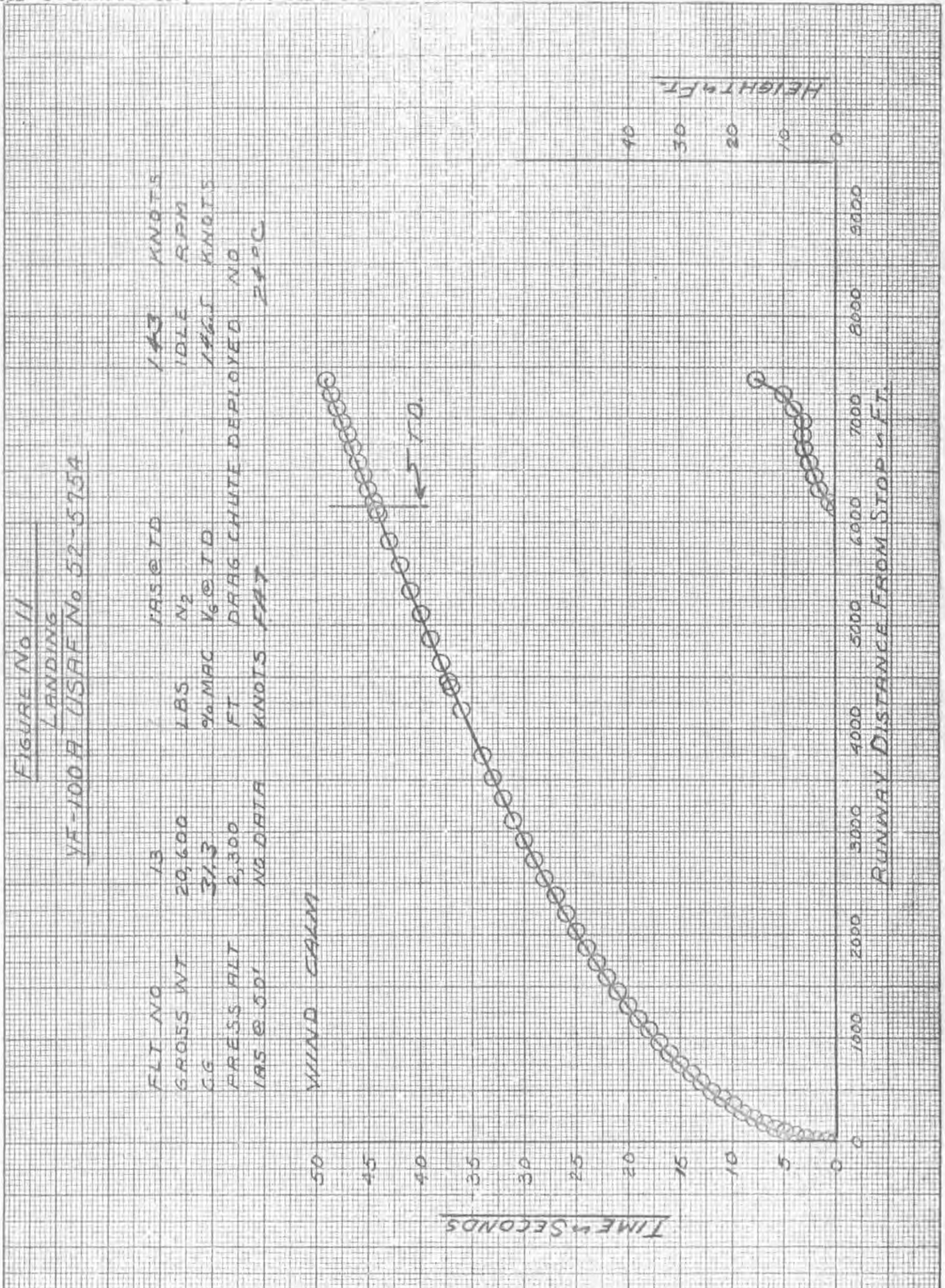
FIGURE No 11
LANDING
YF-100A USAF No 52-5754
FLT NO 13
GROSS WT 20,600 LBS
CG 31.3 % MAC
PRESS ALT 2,300 FT
IAS @ 50' NO DATA KNOTS
IAS @ TD 143 KNOTS
N2 IDLE RPM
Vs @ TD 146.5 KNOTS
DRAG CHUTE DEPLOYED NO
FAT
WIND CALM
T.D.
TIME ~ SECONDS
HEIGHT ~ FT
RUNWAY DISTANCE FROM STOP ~ FT

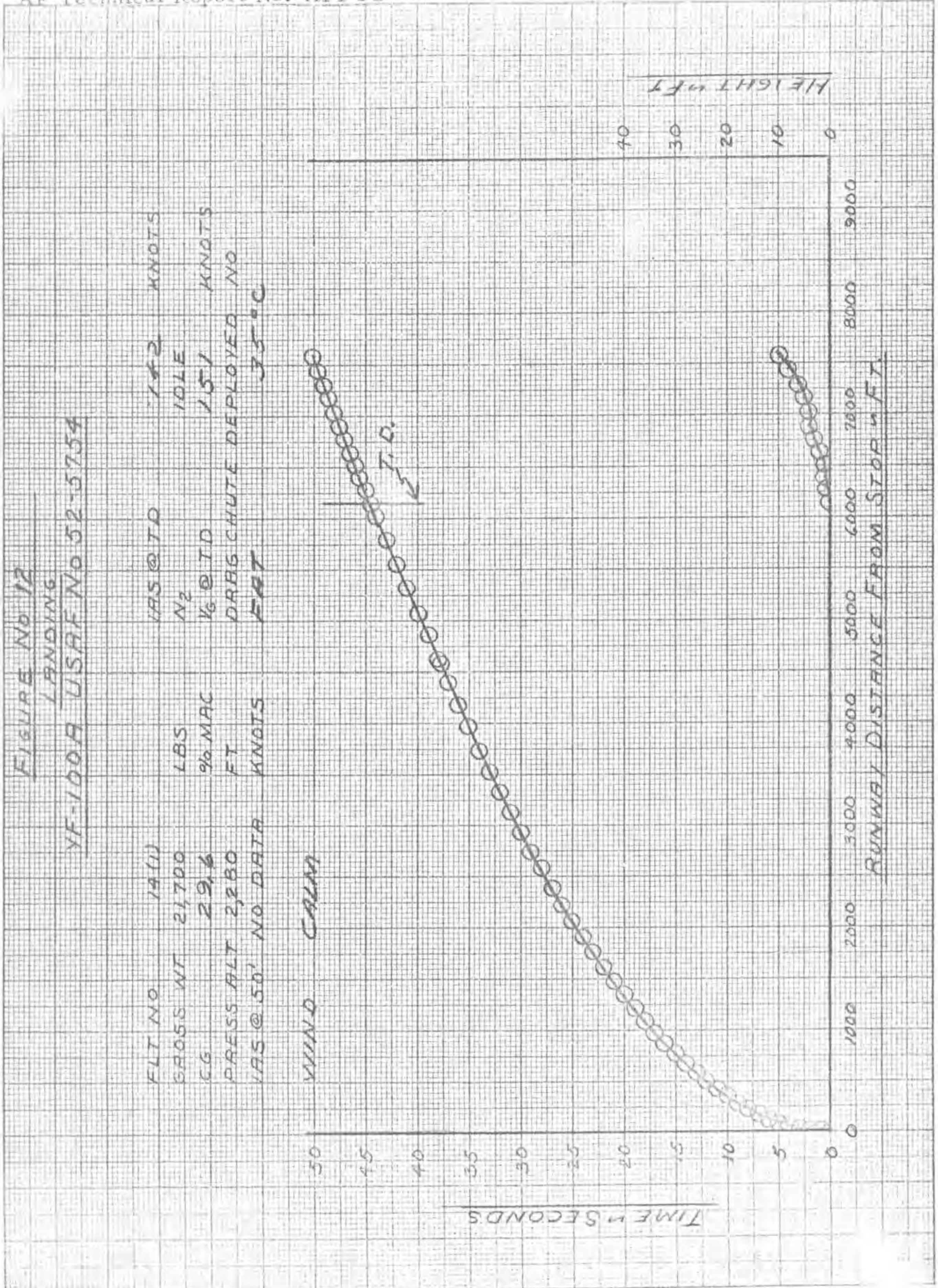
FIGURE No 12
LANDING
YF-100A USAF No 52-5754
FLT NO 14(1)
GROSS WT 21,700 LBS
CG 29.6 % MAC
PRESS ALT 2,280 FT
IAS @ 50' NO DATA KNOTS
IAS @ TD 142 KNOTS
N2 IDLE
Vc @ TD 151 KNOTS
DRAG CHUTE DEPLOYED NO
FAT 35 °C
WIND CALM
T.D.
TIME ~ SECONDS
0 5 10 15 20 25 30 35 40 45 50
HEIGHT ~ FT
0 10 20 30 40
RUNWAY DISTANCE FROM STOP ~ FT
0 1000 2000 3000 4000 5000 6000 7000 8000 9000

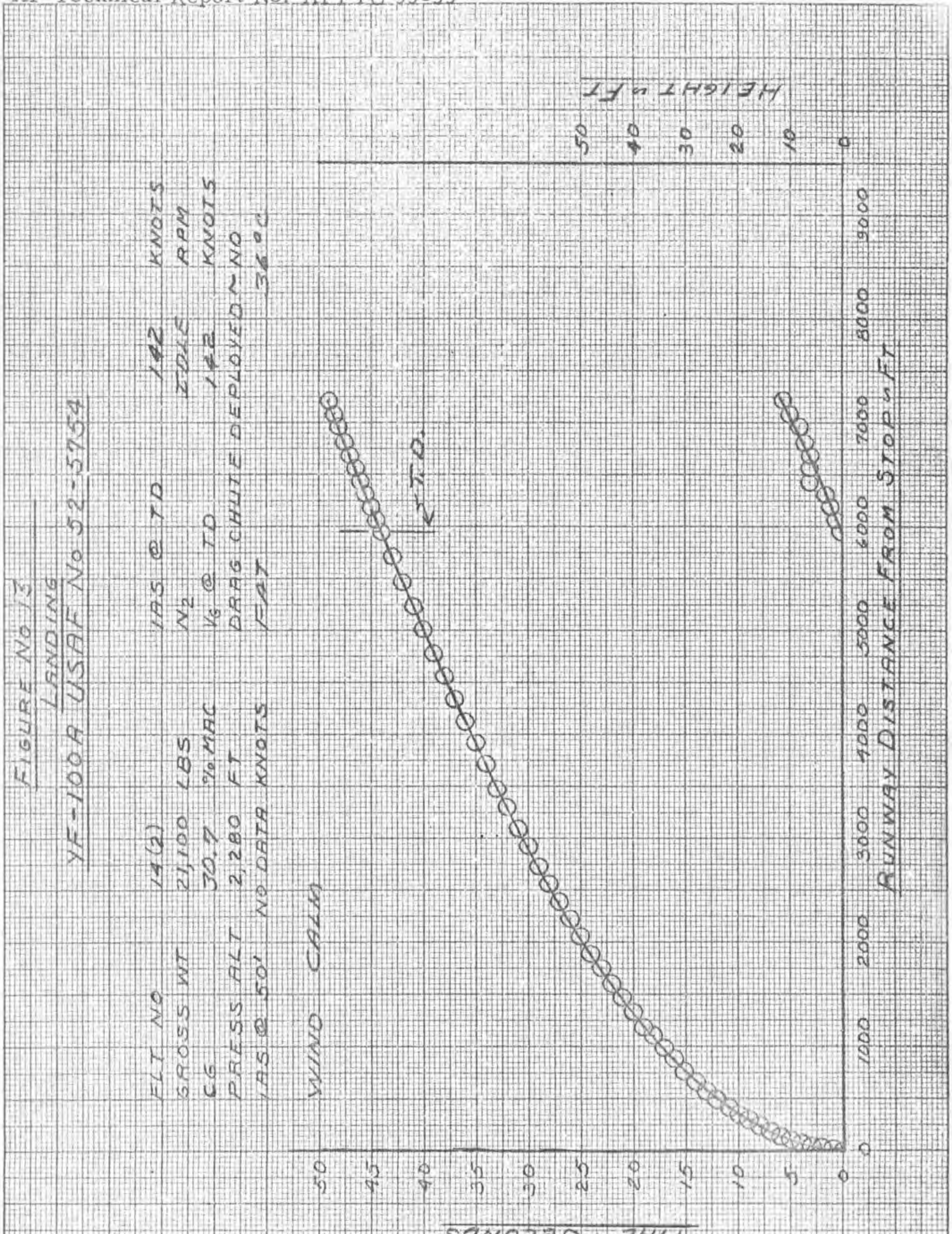
FIGURE NO 13
LANDING
YF-100A USAF No 52-5754
FLT NO 14(2)
GROSS WT 21,100 LBS
CG 30.7 % MAC
PRESS ALT 2,280 FT
IAS @ 50' NO DATA KNOTS
IAS @ TD 142 KNOTS
N2 IDLE RPM
Vg @ TD 142 KNOTS
DRAG CHUTE DEPLOYED ~ NO
FAT 36°C
WIND CALM
T.D.
TIME ~ SECONDS
HEIGHT ~ FT
RUNWAY DISTANCE FROM STOP ~ FT
0
1000
2000
3000
4000
5000
6000
7000
8000
9000
0
5
10
15
20
25
30
35
40
45
50
0
10
20
30
40
50

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